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EXPERIMENTAL TECHNIQUES FOR SUPERSONIC COMBUSTION RESEARCH IN A SHOCK TUNNEL

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SUMMARY

The flight corridor for scramjet operation extends to about Mach 20 at altitudes up to 50,000 metres. Simulation of the combustion chamber conditions for research and development therefore requires air at extremely high enthalpy and pressure. The tailored interface shock tunnel is an economic test facility which can attain the required conditions, despite the short testing time. The supersonic combustion test chamber may be directly connected to the shock tunnel nozzle. Starting of the flow in the test section is accompanied by a transient shock whose strength may be reduced by pre-evacuation. Fuel (usually hydrogen) is injected into the test section and the mixing, reaction and aerodynamic processes are investigated.

In addition to conventional pressure and shock speed instrumentation, the feasibility of the measurement of gas velocity by electromagnetic induction has been studied and shown to be a promising technique at the higher flight speeds. The electrical conductivity of the gas limits the range of applicability and radio frequency probes have been used to indicate its magnitude. High speed gas sampling valves may be used to measure the fuel concentration in the mixing region, however care must be taken

with probe design. The proven value of flow visualization techniques may be attained by high speed cine photography.

1.0 INTRODUCTION

At the present time, the scramjet (supersonic combustion ramjet) appears to be the most promising power plant for hypersonic propulsion. Extensive performance calculations[1,2] have been carried out based on hypothetical efficiency of the intake, combustor, nozzle and vehicle, however experimental investigation of these components is still at an early stage. Although a complete understanding of aerodynamics and chemical kinetics would permit confident predictions of component behaviour, such an objective is too vast, and it is necessary to confine attention to the actual conditions likely to be encountered. Experimental research programmes must therefore be closely associated with the hypersonic mission to be achieved if the accumulation of irrelevant data is to be avoided.

It is convenient to consider first the 'flight corridor' (Fig. 1 (a)) of a scramjet powered vehicle, since this illustrates the range of stagnation pressure and stagnation temperature to be considered. These parameters are plotted on Figs. 1 (b) and (c) for both ideal air ($\gamma = 1.4$) and real air (equilibrium), and it can be seen that real gas effects must be included in this speed range when stagnation conditions are required. For cruise missions a maximum speed of about Mach 15 appears likely[3], whilst for boost acceleration missions, speeds up to Mach 20 are conceivable. As shown in Fig. 1 (a), cruise vehicles would

also operate at lower dynamic pressure (higher altitude) than boost vehicles, and we are therefore interested in flight stagnation pressures and temperatures up to 5000 Bars, 8000°K. and 10^5 Bars, 14000°K. for the cruise and boost missions respectively.

1.1 COMBUSTION CHAMBER CONDITIONS

In this particular study we are concerned with the combustion chamber conditions, rather than the ambient conditions, and an air stream corresponding to that at the exit of an intake is therefore required for experimental research. Practical intakes do not diffuse the air with complete efficiency and the intake efficiency decreases with the amount of diffusion required. In general, two parameters are required to specify the amount of diffusion and efficiency of diffusion, and in this study velocity ratio (V_3/V_1) and diffusion factor (M_1/M_3) will be used to define the former, whilst kinetic energy efficiency (η_D) and process efficiency (K_D) will be used to define the latter. (Reference stations through the engine are illustrated on Fig. 2). Although the velocity ratio (V_3/V_1) and kinetic energy efficiency (η_D) are the more obvious intake performance parameters, the Mach number ratio (M_1/M_3) and process efficiency (K_D) remain more nearly constant (for scramjet engines) as flight Mach number and amount of diffusion are varied. (As shown in References 3 and 4 the optimum value of M_1/M_3 is approximately 3 and the likely value of K_D is approximately 0.9 in the hypersonic speed range).

The static pressure (p_3) and temperature (t_3) at the entry to a scramjet combustor therefore depend on the flight Mach number (M_1), dynamic pressure (q), diffusion factor (M_1/M_3) and process efficiency (K_D) of the intake. It will be shown that the temperature (t_3) is normally below 2000°K. and therefore ideal gas relations can be used to illustrate the relationship between these parameters with little loss in accuracy.

Considering first the static temperature (t_3), it is apparent that for ideal gas this is independent of the intake efficiency, thus:-

$$\frac{t_3}{t_1} \approx \frac{h_3}{h_1} = 1 + \frac{V_1^2}{2Jh_1} \left\{ 1 - \frac{V_3^2}{V_1^2} \right\} \qquad (1)$$

or in terms of the diffusion factor:-

$$\frac{t_3}{t_1} = \left\{ \frac{1}{M_1^2} + \frac{\gamma - 1}{2} \left(\frac{M_1}{M_3} \right)^2 \right\} \Big/ \left\{ \frac{1}{M_1^2} + \frac{\gamma - 1}{2} \right\} \qquad (2)$$

These functions are plotted on Fig. 3 (a) for the hypersonic speed range, and it can be seen that the velocity in the combustion chamber is only a few percent below its free stream value, especially at the higher hypersonic speeds. Since the standard ambient temperature (t_1) is defined by the ARDC atmosphere, the value of the combustion chamber entry temperature (t_3) can be plotted as a function of the flight Mach number (M_1) for any given value of dynamic pressure. In figure 3 (b), t_3 is plotted

against M_1 for q = 2 Bars, and a range of intake diffusion factors. Taking $M_1/M_3 \sim 3$ it is apparent that the combustion chamber entry temperature lies between 800°K. and 2000°K. in this speed range, with the higher temperatures being associated with the higher flight speeds. The small effect of dynamic pressure on t_3 is illustrated on Fig. 3 (c) for q = 0.1, 0.5 and 2 Bars.

Turning now to the pressure in the combustion chamber, this is given by:-

$$\frac{p_3}{p_1} = \left[\frac{1 + \dfrac{V_1^2}{2Jh_1} \left(1 - \dfrac{V_3^2}{V_1^2}\right)}{1 + \dfrac{V_1^2}{2Jh_1} (1 - \eta_D)} \right]^{\frac{\gamma}{\gamma - 1}} \tag{3}$$

or in terms of the diffusion factor and process efficiency:-
(where $\eta_D = K_D + (1 - K_D)(V_3/V_1)^2$ see Ref. 5).

$$\frac{p_3}{p_1} = \left[1 - K_D \left\{1 - \frac{M_3^2}{M_1^2}\right\} \middle/ \left\{1 + \frac{2}{(\gamma - 1) M_1^2}\right\}\right]^{\frac{-\gamma}{\gamma - 1}} \tag{4}$$

This latter function is plotted in Fig. 4 (a) for the relevant range of K_D and M_1/M_3, and it can be seen that the static pressure is normally increased by about two orders of magnitude in the intake. Again the Mach number/altitude relationship given by particular values of the dynamic pressure can be used to obtain the absolute value of the parameter - in this case combustor

entrance pressure. Taking $q = 0.1$, 0.5 and 2 Bars at $K_D = 0.9$ and $M_1/M_3 = 3$, the characteristics plotted in Fig. 4 (b) are obtained and it is apparent that the p_3 range of interest lies between 0.1 and 5 Bars.

In a flight scramjet these pressures, temperatures and Mach numbers at entry to the combustor are derived from the flight stagnation conditions by non-isentropic diffusion in the intake. In the ground test facility, isentropic expansion from a stagnation reservoir can be obtained in a Laval nozzle designed for the appropriate Mach number. Thus the ground test facility only needs to provide a stagnation pressure P_3 which will be considerably lower than P_1, however the stagnation enthalpy must be the same in both systems for complete simulation. Although real gas effects become more significant as the stagnation conditions are approached, the magnitude of the reduction in stagnation pressure in the intake can be conveniently illustrated by ideal gas relations thus:-

$$\frac{P_3}{P_1} = \left[1 + (1 - K_D)\left\{\frac{h_3}{h_-} - 1\right\}\right]^{\frac{-\gamma}{\gamma - 1}}$$

$$= \left[1 + (1 - K_D)\left(1 - \frac{M_3^2}{M_1^2}\right)\Big/\left\{\frac{2}{(\gamma - 1)M_1^2} + \frac{M_3^2}{M_1^2}\right\}\right]^{\frac{-\gamma}{\gamma - 1}} \qquad (5)$$

This function is plotted in Fig. 5 for a range of K_D and M_1/M_3, and it is shown that the stagnation pressure in the ground facility is about an order of magnitude less than the equivalent flight case. As can be seen by reference to Figures 1 (b) and 1 (c),

this still represents extremely high pressures (well in excess of 1 Kilobar at the higher hypersonic speeds) however this reduction in stagnation pressure is a welcome feature, and means that it is much more practicable to carry out realistic 'connected' tests of a scramjet combustor than tests of an engine complete with intake.

1.2 SCALING PARAMETERS

The above discussion shows that complete simulation of all gas dynamic parameters in a scramjet combustor is an arduous task for any facility, and we must consider which scaling parameters are likely to be most significant so that some of the requirements may be relaxed. The important phenomena to be studied are the turbulent mixing of fuel and air, with simultaneous chemical reaction, and the interaction of the resulting pressure gradients with the duct geometry.

a) The mixing is governed by the velocity and density ratios, Reynolds numbers, and the boundary layer at the fuel jet entry.

b) The chemical reactions are governed by the local pressure, temperature, concentration and mixedness of the fuel and air throughout the reaction zone.

c) The interaction between area changes in the duct and the heat addition process can be very large, especially at transonic flow conditions and near combustion limits.

To study all these factors simultaneously, it is necessary that the pressures, temperatures, velocities and

dimensions of the fuel and air streams should be correctly simulated. Of course facilities which study only one factor (e.g. mixing, chemical kinetics, or aerodynamics) have been used but we now wish to study the integrated effect, and little compromise is possible.

Fortunately, sufficient is now known about supersonic combustors that the critical regions can be identified and it is found that the major problems can be studied without maximum test facility capability (i.e. 10^5 Bars at 14000°K). Considering first the higher hypersonic speeds, it is shown in Ref. 6 that the combustion time is sufficiently short (less than 100μ sec.) so that the system is mixing controlled. The pressure and temperature can then be reduced with little loss in accuracy since the mixing depends predominantly on velocities and densities. In addition, at these speeds a large part of the stagnation energy is associated with the velocity, and a small reduction in velocity greatly reduces the required stagnation conditions. It is therefore concluded that for flight speeds between Mach 10 and Mach 20, connected combustion testing can be adequately covered by stagnation temperatures of 5000°K. and stagnation pressures of 200 Bars. If however an intake is to be included, as in complete engine tests, then the Mach Number in the intake is important and the reduction in temperature must be carefully matched to the reduction in velocity so that the Mach number is unchanged. In this case at $M_1 = 20$ a combustion chamber temperature of 1500°K.

would correspond to a stagnation temperature of ~ 8000°K. (real gas), which is much more readily attainable than 14,000°K. required for true simulation. The corresponding stagnation pressure is approximately 10^5 Bars at $M_1 = 20$, however this can be reduced if higher stagnation temperatures are available. (This follows because the overall combustion time is a function of both pressure and temperature, and a reduction in combustor static pressure can be compensated by an increase in static temperature).

Considering next the intermediate flight speeds between Mach 7 and Mach 10, at which the supersonic combustion is usually controlled by the chemical kinetics. In this case the Mach number in the combustor is not a critical parameter, whilst temperature, pressure, mixture strength and residence time are important. The stagnation conditions can therefore be reduced significantly provided the mixing is simulated by retaining the correct velocity relationship between the two streams[7]. A stagnation temperature of ~2200°K. is adequate to study this region.

Finally we must consider the lower flight speeds (below Mach 7) when the combustion chamber is operating transonically. It is shown in Ref. 8 that the critical condition of thermal choking, which is $M_5 = 1$ in a constant area duct, assumes other values for non-constant area ducts. It is essential therefore that the Mach number be correctly simulated if this limit is approached. This phenomena is particularly important below a flight Mach number of

7 - which is also the region at which the combustor static temperature is too low for spontaneous ignition of the fuel, and some form of recirculation or piloting[9] is necessary to stabilize the flame. The spread of flame from a pilot is only slightly affected by the static temperature of the main stream, hence true temperature simulation is not too critical in this region, and indeed some transonic combustion tests have been carried out with no air preheating[10].

1.3 FACILITIES

From the foregoing it is apparent that supersonic combustors for the different flight regimes can be conveniently studied with different facilities. Conventional supersonic wind tunnel systems (modified to handle the potentially explosive exhaust), with vitiation or heat exchanger preheating up to say 1000°K., can be used for studies up to $M_1 = 7$.

Ceramic preheaters (e.g. pebble beds) or arc heaters can give temperatures above 2200°K. and are therefore useful to cover the intermediate range between $M_1 = 7$ and $M_1 = 10$.

The higher speed range from Mach 10 to Mach 20 requires minimum stagnation conditions of 5000°K at 200 Bars for connected tests, and up to 8000°K. at 10^5 Bars for complete engine tests. These pressures and temperatures are beyond those attainable by any present day continuous ground test facility, and intermittent facilities must be considered. It would, however, be most valuable if the arc heated wind tunnel could be developed to this capability,

although free flight tests using a rocket boosted test vehicle will offer a cheaper alternative in the immediate future.

Two types of intermittent facility can attain the required conditions - the hot shot tunnel and the reflected shock tunnel. Current hot shot tunnels are expensive to build since they need considerable electrical energy storage, and also suffer from the disadvantage that the oxygen in the air is depleted during heating. It is anticipated that these disadvantages can be largely overcome, however the reflected shock tunnel appears to be an attractive alternative at the present time.

This study will therefore be concerned with the application of a 15 cms (6") shock tunnel built at Sheffield University, to the study of supersonic combustion for scramjet operation between Mach 10 and Mach 20. A general view of this facility is given in Plate 1.

Although connected test techniques are being used at present with this facility, complete engine tests will eventually be essential. This arises because the velocity profiles at the exit of a real scramjet intake are likely to be very non-uniform, since different streamlines are processed by vastly different shock and boundary layer conditions. It may be possible to extend the applicability of connected testing by the design of nozzles which give deliberately non-uniform flow, thus duplicating the intake diffuser exit flow determined independently on a

hypersonic wind tunnel.

2.0 THE SHOCK TUNNEL

Many references (e.g. 11) detail the design of shock tunnels, and only the points relevant to supersonic combustor testing will be discussed here.

The main disadvantage of the shock tunnel arises from the short testing time of a few milliseconds. The important point is that this time should be long compared to the important combustor processes. In the flight speed range being considered, Figures 3 and 4 show that the intake velocity ratio V_3/V_1 is about 0.9, hence the air velocity in the combustor is greater than $\sim$3000 m/s. In Ref. 3 it is shown that the fuel injection velocity will be about half the air velocity, i.e. $\sim$1500 m/s, hence all the gases will flow through a combustor of 1 metre length in less than 0.75 milliseconds. Testing times in excess of one millisecond are therefore adequate to study the combustion process.

The tailored interface mode of operation of the reflected shock tunnel gives the longest running time for a given length of driven tube, and the capability of such a facility is shown on Figs. 6 (a) and (b) as a function of primary shock Mach number[12]. It can be seen that air stagnation temperatures of $\sim$5000°K. and 8000°K. required for direct connected and complete engine tests, can be produced with cold hydrogen, hot hydrogen and

hot helium driver gases. The corresponding testing time varies from 0.3 to 0.15 m.sec/metre and hence a driven tube length in excess of 7 metres is required. A great excess in length can result in shock attenuation problems, and a length of 8.7 metres is used in the supersonic combustion research shock tunnel at Sheffield University.

With cold hydrogen driving, the diaphragm is simply burst by pressure. The hot helium driving follows the technique developed by General Electric(13), in which a mixture of 70% helium and stoichiometric hydrogen: oxygen, is ignited by eight high energy discharge igniters (each 4μF at 1500 volts). The combustion results in a smooth six to seven fold pressure rise over a period of about 20 m.sec., at which time the diaphragm opens. The required driver pressure is usually about 20% greater than the final air stagnation pressure. A detailed description of the facility design and calibration is given by M.P. Wood(12).

It is interesting to note that hot hydrogen driving would be very suitable for scramjet testing, and in view of the high cost of helium, a rapid electrical hydrogen heating technique (in which the driver tube is cooled by thermal inertia) could be well worth developing. The alternative of combustion heating of hydrogen (burning some hydrogen with oxygen) can result in undesirable detonations, and indeed less than 70% helium mixtures have produced very rough combustion in our facility.

In addition to the familiar limitation of testing time due to the arrival at the nozzle of the tail or reflected head of the expansion from the diaphragm, recent studies have shown that contact surface interactions in the driven tube may also curtail the useful running period. This process has been analysed by L. Davies in Ref. 14 where it is shown that two phenomena are responsible, namely contact surface instability and shock bifurcation caused by shock interaction with the boundary layer. This latter process results in annular jets of driver gas penetrating the shock heated gas in the stagnation region and diluting the test gas flowing through the nozzle. The onset of these two processes usually occur at similar shock Mach numbers, and calculated critical shock Mach numbers for the two driver systems in current use at Sheffield are shown in Fig. 7 (a). Fortunately these limits are above the tailoring Mach numbers, however experiments are in hand to investigate this problem and will be reported in due course.

Since the stagnation temperatures are so high, the air is dissociated a few percent, and the possibility of the flow freezing in the nozzle must be considered. It is well known that the flow freezes in hypersonic nozzles (with an area ratio of about 10^3) however in this study, area ratios are approximately 10 and the test section static pressures and temperatures are still comparatively high. The freezing point in a hypersonic

nozzle has been analysed by Bray[15] based on results calculated at Cornell Aeronautical Laboratory. Data from this reference is plotted in Fig. 7 (b) together with the combustion chamber conditions corresponding to a range of flight Mach numbers, dynamic pressures and diffusion factors. It can be seen that the flow is frozen for all conditions, hence the translational temperature will be lower than would be obtained with equilibrium flow. However it will be noted that the freezing only occurs at temperatures just above those being considered, so that the effects of freezing will be slight. In general it is expected that freezing effects will increase the oxygen atom level above its equilibrium value at entry to the combustion zone.

3.0 TEST SECTION

3.1 FLOW STARTING

The design of a connected combustion test section poses some interesting problems compared to the hypersonic shock tunnel since the test section to throat area ratios are much smaller and the test section static pressure is usually much greater. In fact with test section static pressures of about 1 Bar (as shown above), it would appear at first sight as though the conventional evacuated dump tank could be dispensed with, and the exhaust discharged direct to atmosphere. Indeed our facility is not fitted with a dump tank and free jet tests at $M_T = 3$ have been successfully conducted with a robust $M = 3$

nozzle. However when a 1 metre long combustion chamber is fitted, the tunnel starting conditions must be considered since high pressures can be built up during the starting process. This phenomena is considered in Refs. 16 and 17 and in this case can be interpreted as follows.

On bursting of the nozzle diaphragm, a shock wave travelling at M^* is initiated at the throat and advances through the nozzle gas followed successively by the contact surface, a backward facing shock and air at the required test conditions. The pressure between the two shocks (p_y) is determined by the initial pressure in the nozzle (p_N) and the shock Mach number (which decreases from M_s^* to M_{ST} as it passes down the nozzle).

For any given stagnation pressure and nozzle area ratio, there is a maximum initial nozzle pressure which will restrict this starting pressure to the required level. Any further decrease in p_N will give a decrease in p_y even though it results in an increase in the starting shock Mach number. As a second requirement, in order that the starting process should be rapid, there exists a minimum value of M_s^* which is a function of the nozzle area ratio as given in Ref. 15. This is interpreted in terms of the combustor test Mach number (M_T) on Fig. 8 (a) and it can be seen that $M_s^*{}_{min}$ increases as the test Mach number increases. The corresponding minimum starting pressure ratio (P_s/p_N) to give this Mach number is plotted against the test

Mach number in Fig. 8 (b), thus the nozzle section should be evacuated to give at least this pressure ratio to ensure rapid starting. At this minimum level, the starting shock Mach number in the test section is given by[18] :-

$$M_{ST} = \left\{ 1 + \frac{(M_s{*}^2 - 1)}{(A_T/A*)^{.4}} \right\}^{\frac{1}{2}} \qquad (6)$$

This value is plotted on Fig. 8 (a) and it can be seen that the minimum value of M_{ST} is almost constant at Mach 5 throughout the test section Mach number range of interest. The corresponding static pressure rise (p_y/p_N) through this shock is approximately 30. Referring again to Fig. 8 (b) it can be seen that the running pressure ratio is approximately an order of magnitude less than the minimum starting pressure ratio, so that if the test section is evacuated to about one tenth to one thirtieth of the test static pressure ($p_T = p_3$), then the starting process will be rapid and the overpressure during the starting shock will be reduced to negligible proportions. Since test section pressures of about 1 Bar are usually required, the connected test shock tunnel configuration does not require such high vacuum as the conventional hypersonic shock tunnel, however some evacuation is essential if the test section is not to be damaged by the starting shock.

Our shock tunnel is therefore fitted with a blast tube 6 metres long (and 0.25 metres diameter) exhausting to atmosphere

through a Melinix diaphragm. This tube is normally evacuated to about 0.1 Bars, and tests have proved this arrangement to be satisfactory and necessary since earlier omission of the vacuum resulted in failure of the test section!

3.2 TEST SECTION DESIGN

The scramjet combustor test section is very similar to a conventional supersonic wind tunnel, having a nozzle to accellerate the flow followed by the test section proper. Due to the high static pressure it is not usually necessary to include a diffuser unless the scramjet exhaust nozzle is included with the combustion chamber, and even in this case it would be better to use a large evacuated dump tank. At present we do not include the exhaust nozzle and the combustor is simply a variable area duct about 1 metre long. The length is determined by the combustor mixing and kinetics which we wish to study, and in common with other supersonic wind tunnels, the length determines the minimum acceptable cross-sectional area (due to friction considerations).

For a scramjet combustor the boundary layers are almost invariably turbulent[3], whilst in the test section, the Reynolds numbers are $\sim 10^6$ and the boundary layer is marginally turbulent. (Transition will occur at a Reynolds number of about 5×10^6). The corresponding mean skin friction coefficient can be taken as ~ 0.0025 and Fig. 9[19] shows that the flow in a constant area

duct will then decrease from Mach 4 to Mach 3 in 11 duct diameters with a corresponding reduction of stagnation pressure to 25% of its initial value. It is therefore essential that the length/diameter ratio of the test section should be as small as possible, especially at the higher test Mach numbers, and a maximum value of 10 is suggested. Since the combustor length is one metre, the minimum useful diameter is approximately 10 cms. at the exit of the shock tunnel nozzle. The nozzle area ratio then determines the throat area as ~ 6.5 cm^2., and since the shock tube driver to throat area ratio should be about 25, the corresponding minimum diameter of the shock tube is ~ 0.145 metres (i.e. approximately 6 inches). When these dimensions are associated with the stagnation pressure capability of $> 10^3$ Bars as described above, the result is a large shock tunnel compared to many University shock tubes, but is nevertheless the minimum practicable size for this particular field of work.

Returning to the friction considerations in a constant area tube, the large effect of friction coefficient is also illustrated in Fig. 9 (a). Although a value of 0.0025 was assumed above, the value varies from about 0.0015 to 0.0035 for laminar and turbulent boundary layers respectively[19], and the length required for a given change in Mach number is inversely proportional to C_f. It may therefore be possible to take advantage of this difference between boundary layers in flight engines and connected model tests by scaling the diameter down in inverse proportion

to the change in friction coefficient, i.e. by a factor of approximately 3. The change in duct static pressure and temperature resulting from the friction is illustrated in Fig. 9 (b), and it can be seen that a factor of two can be readily induced. The large effect of such a change on the combustion kinetics will be appreciated, as will the fact that measurements of wall static pressure can be interpreted as due to heat release or friction effects. Experimental results can not therefore be analysed in terms of one dimensional inviscid flow, and detailed knowledge of the flow field is necessary.

Although a constant area test section has been discussed above for simplicity, practical combustors will usually vary in area to produce, for example, combustion at constant pressure, constant Mach number, etc. The required variation of pressure with flow area is often specified by the Crocco relation[8,20]:

$$\frac{p}{p_3} = \left\{ \frac{A}{A_3} \right\}^{-\varepsilon/(1 - \varepsilon)} \tag{7}$$

and the area of the combustor should be variable along its length to constitute a versatile research apparatus. This variation is most readily accomplished by the use of a two dimensional test section with flexible walls supported on screw jacks. If the side walls are transparent, conventional optical techniques can be used for flow visualization and temperature

measurements. The two dimensional geometry is also suitable for the essential flow profile measurements, as discussed above. A diagram of a basic Mach 4 test section is given in Fig. 10, together with a photograph of the nozzle section in Plate 2.

4.0 FUEL INJECTION

The supersonic combustion process can be investigated by either small disturbance methods or near stoichiometric fuel injection. In the former method, the principle of equivalence between heat, mass and volume sources can be invoked to show that a small fuel jet, even with combustion, will not greatly alter the flow pattern in the test area. The fuel jet flow can then be analysed by the usual methods employed for jets in an infinite stream, and compared with the experimental results of flame location, temperature distribution and concentration distribution. Plate 3 shows such an injector mounted through the throat of a Mach 3 nozzle, together with the nozzle expansion bell.

With near stoichiometric fuel injection, pressure temperature and velocity changes are produced throughout the flow field and a much more elaborate analysis is required. In general the fuel injector configuration will be complex, possibly involving multiple vortices to promote rapid mixing, and a semi-empirical theoretical mixing analysis must be used. The duct area change at the fuel injection station will often be significant, for example Ref. 3 shows that for stoichiometric hydrogen fuel,

the fuel jet area may be as much as 30% of the duct area. The engine specific impulse which results from downstream fuel injection is a significant part of the total impulse at the higher hypersonic speeds and it is therefore likely that such a geometry will be used in practice. In the shock tunnel test section, it is convenient to study this geometry by wall slot injection along the upper and/or lower walls, thus giving a two dimensional flow field. The instrumentation described below can be used to determine the results from such a test, bearing in mind that boundary layers, friction and area change can have a profound effect on the flow pattern. In general, shock tunnel tests are conducted with the walls at room temperature, however in a flight engine the walls will be hot and therefore the heat transfer rate reduced. Since there is a significant effect of wall heating on boundary layer profiles and transitions, as test techniques become more sophisticated it may be necessary to preheat the test section walls electrically before firing the tunnel.

When fuel is to be introduced into the test section, the timing of injection must be carefully synchronized with firing. Early injection results in depletion of the vacuum, so that the time must be controlled to about 0.5 millisecs. This may be achieved by fast acting solenoid valves which require about 1 millisecond to open, however an alternative system consists of a multiple sheet plastic diaphragm which is removed by passing

a large current through a wire introduced between the sheets. This technique was developed at GASL[21] for supersonic combustion research using a 3" shock tunnel.

The flow rate of the hydrogen is easily controlled by an orifice connected to a small reservoir of a few hundred millilitres capacity. This reservoir is precharged to a known pressure, and following opening of the valve will discharge exponentially in a period of about one second. The pressure, and hence flow rate, remains essentially constant during the tunnel running time, and is readily monitored by means of a pressure transducer. If heated fuel is required, the reservoir can be placed in a furnace, and the ducts to the test section heated.

5.0 STANDARD INSTRUMENTATION OF THE SHOCK TUNNEL

In most shock tunnels used in supersonic combustion research an evaluation of the test section gasdynamic conditions is found by measuring only two properties downstream of the nozzle. These are usually temperature and pressure, which when combined with a knowledge of the stagnation conditions can be used to calculate all the physical properties of the test gas. The use of a Mollier Chart, or real gas data in some form, is therefore required.

The monitoring of temperature is accomplished by using a double-beam reversal technique[22]. The spectral lines of sodium are usually employed although the chromium and cesium

lines are equally well suited. Fig. 11 is a diagram of the equipment used to measure the temperature on the Shock Tunnel at Sheffield, based on the sodium doublet. Sodium is introduced into the tunnel in the form of sodium chloride. It is preferably to seed the test gas with a smoke of a sodium compound but the above method does not seem to suffer from a lack of sodium vapour. Quartz iodine lamps are now used as these give a higher brightness temperature than the tungsten filament lamps formally used(12). The vibrational and translational temperatures are assumed to be in complete equilibrium so that the temperature measured by this method (the vibrational temperature) is equal to the true gas temperature.

Piezo-electric transducers are used to measure the pressure levels throughout the shock tunnel. In the test section two separate methods of measurement are used. The wall static pressures are found by Kistler 701 transducers set in anti-vibration mounts, and exposed to the flow via small holes. Alternatively a sting arrangement permits the use of a static or pitot pressure probe built to a National Physical Laboratory design(11). The signals from the charge amplifiers are usually filtered to reduce the noise level from mechanical vibrations, despite precautions taken to avoid this.

Two alternative mechanisms are used to measure the primary shock speed depending on whether the tunnel is driven by combustion heated helium or cold hydrogen. For combustion

driving with the primary shock speed $M_s > 10$ ionisation gauges are used. These consist of a central electrode held at 300 volts surrounded by an earthed electrode. On passage of the shock, the gap breaks down and a pulse may be obtained. At lower shock speeds pressure transducers are used because of the low level of ionisation present.

6.0 THE MEASUREMENT OF GAS VELOCITY BY ELECTROMAGNETIC INDUCTION

The possibility of measuring velocity by electromagnetic induction has been known for a long time and commercial instruments are available to meter liquid flows. The application of the method to metering ionised gas flows has not received so much attention, and this study presents the application of this technique to velocity measurement in the supersonic combustion test section. The test gas after expansion will be ionised and therefore the method should be applicable to measuring its velocity. If this test gas is passed between the poles of a magnet so that the magnetic field is perpendicular to the velocity vector then the magnitude of the induced voltage is given by:

$$v = B\, V_m\, D \text{ volts} \qquad \ldots\ldots \qquad (8)$$

This equation gives the magnitude of the measured voltage only if all the walls of the test section are non conducting[23], if this is not so then the observed signal will be less than that given by Eq. (8). (The ratio of the observed signal to the theoretical signal being defined as the sensitivity).

Early workers in the field[24] experienced difficulty due to the appearance of a voltage when there was no imposed magnetic field. This voltage was of the same order of magnitude as the induced voltage (c 1 volt) and made interpretation of the results difficult. Holbeche[25] investigated this phenomena in a 2 inch shock tube, and found that this voltage could be of either polarity. It therefore seems necessary to have the induced voltage as large as possible and also to use a different measuring circuit.

Cason[26] reports the use of the technique for the measurement of velocity in a plasma jet. Results were found to be in agreement with calculations although no direct comparison with other forms of velometer was reported. The results using a d.c. excited magnet were found to be affected by thermionic emmission at the electrode surfaces and an a.c. excited magnet was substituted to eliminate such effects. A shock-tube induction flowmeter is described by Croce[27] but once again the results given are only compared with theoretical induced voltage values although the sensitivity of this type of instrument can be over 100%[28]. Rolls-Royce Ltd.[29] have also measured velocities with an induction system during studies on ethylene-oxygen detonations. By comparison with drum camera measurements they found that it was necessary to apply a calibration factor of between 1.5 and 1.7 to the velocity predicted from the induced voltage.

6.1 CONDUCTIVITY OF THE TEST GAS

Equation 8 predicts the induced voltage provided that

the fluid has a finite value of electrical conductivity. The level of the conductivity in the test section will determine the external impedance necessary to accurately determine the induced voltage. If a 10% loss in uncorrected voltage is acceptable, then the equivalent circuit yields $R_L \geqslant 9R_g$ where R_L is the load resistance and R_g the gas resistance. Even if σ (the gas conductivity) is known, the determination of R_g presents some difficulty, because only when $A >> D^2$ (where A is the area of the electrodes) is it equal to $D/A\sigma$. The technique used by Pain and Smy[30] was therefore adopted.

A small perspex box was constructed, with the same internal dimensions as the test section, and with provision for placing a pair of electrodes in the side walls. The box was filled with salt solution of known conductivity (σ') and the resistance between the electrodes (R_g') found using an a.c. bridge circuit, then $R_g = R_g'\sigma'/\sigma$. Using flush electrodes made of 1.59 mm. diameter silver steel rods, it was found that $R_g = 10^3/\sigma$. All types of electrode can be tested in this way before fitting to the test section.

The expression for R_g is a function of σ, the gas conductivity and some estimate of its value is needed. Viegas and Peng[31] have calculated the conductivity of thermally ionised air above 3000°K. for various pressures. Very little attempt has been made to find conductivities below this level of temperature as the degree of ionisation becomes so small. Coombe[32] gives an

expression for the conductivity of a slightly ionised gas as:-

$$\sigma = \frac{7.05 \times 10^{-13}\, T^{3/4}}{\alpha\, P^{\frac{1}{2}} \exp\,(eV_{i/2kT})} \qquad (9)$$

where V_i is the ionisation potential of the gas and α an appropriate average for the electron-atom collison cross-section. Using this equation it has been possible to extend the data given in Ref. 12 down to a temperature of 1500°K. Fig. 12 shows the result of these calculations for three pressures. The graph shows the effect of the exponential term in eq. (9) and it can be seen that the resistance of the gas will be very high over most of the range of static temperature of interest.

The conductivity of the gas is therefore likely to be in the order of 10^{-8} mho/m so that the resistance between the flush electrodes will be $\sim 10^{11}$ ohms. The measuring circuit must therefore have an impedance of at least 10^{12} ohms unless either an estimate of signal loss can be made or the conductivity increased.. Using a 100:1 or 1000:1 attenuating circuit the input impedance of a sensitive Oscilloscope can be raised to $10^{8} \rightarrow 10^{9}$ ohms. For higher values of input impedance an amplifier based on an electrometer valve must be used.

Seeding of the test gas with materials of low ionisation potential such as caesium or potassium can increase the conductivity by several orders of magnitude. Frost[33] has calculated the

conductivity of various seeded atmospheres and the results for argon can be compared with the results of Lim, Resler and Kantrowitz[34] despite the pressure differences. Seeding with 0.02 atm K., increases the conductivity 10^3 times. Since the same improvement can be expected with air then the limit on the external measuring impedance will be lowered to $\sim 10^9$ ohms. This arises because the air is only very weakly ionised and the majority of the ionisation comes from the seed[33], and is therefore generally independent of the carrier gas. An exception arises when the gas contains OH which strongly absorbs electrons, hence difficulties are anticipated when the electromagnetic velocity meter is used in supersonic hydrogen flames. On the other hand hydrocarbon flames are electrically conducting due to the reactions:

$$CH + O \rightarrow CHO^+ + e$$
$$CHO^+ + H_2O \rightarrow H_3O^+ + CO$$

which produce electrons and this may provide an alternative source of conductivity.

6.2 RANGE OF APPLICABILITY OF THE METHOD

The conditions within the test section must correspond to the simulated conditions within a scramjet combustion chamber. These conditions have been discussed above and using these results the limits of applicability of an induction meter can be found. For a specific flight Mach number, the combustor conditions at

any altitude can be found from Figs. 3 and 4 and hence the conductivity found from Fig. 12. Within the range $M_1 = 5$ to 20 and with $K_D = 0.9$, $M_1/M_3 = 3, 4$ the combustor pressure will lie between 10 and 0.1 bars, hence conductivity will not be greatly affected by pressure within this range.

From the previous discussions it appears that for a measuring circuit of impedance 10^{12} ohms and the resistance between electrodes equal to $10^3/\sigma$, the lower limit of applicability lies between $M_1 \sim 7.0$ and $M_1 \sim 11$ depending on M_1/M_3. This limit is illustrated on Fig. 13. The limit may be further reduced by increasing the external impedance or reducing the internal impedance by (a) different electrode structure or (b) seeding of the gas. Seeding is easily accomplished in the shock tunnel but is impracticable in the scramjet. It is likely that seeding can reduce the limit to $M_1 \sim 5$ when $M_1/M_3 = 4$. This would bring the electromagnetic velocity measuring technique within the range of some continuous facilities such as the pebble bed heated hypersonic wind tunnel.

6.3 ELECTRODE GEOMETRY

Shercliffe(23) has examined many types of electrode configurations to give some estimate of the sensitivity to be expected from any particular arrangement. Theoretical calibration is practically impossible and all types of meter need empirical calibration. One of the simplest cases is that of a channel with non-conducting walls and internal wetted electrodes. This

geometry is also very suitable for supersonic application as no obstacle is presented to the flow. Such a meter can have a sensitivity greater than 1 depending on the aspect ratio (a/b) and on the velocity profile within the duct. In a shock tunnel test section with a contoured nozzle the velocity profile should be reasonably flat hence reducing the sensitivity to near unity. Growth of the boundary layer in the duct may have adverse effects on the voltage recorded by wall electrodes and it may be necessary to use projecting electrodes. The possibility of using a pair of closely spaced electrodes to explore the duct velocity profile can also be examined.

In order to overcome the electrode effects present it will be necessary to use a large value of B. The maximum value of B attainable will therefore limit the extent to which D may be reduced.

At the edges of the magnetic field the induced voltage will decrease and hence cause internal currents to flow. This means that the electrodes must be sufficiently remote from the edges of the field to eliminate these end effects. Shercliffe[35] has analysed the two-dimensional case with an abrupt field and with a fringed magnetic field and finds that for a 1% loss in signal, the ratio of length of steady magnetic field to the interelectrode distance must be at least 3.04. Current leakage through the Hartmann boundary layer is negligible with the ionised gas flows being considered, because the distance taken for such

a boundary layer to form is many times greater than the length of magnetic fields likely to be used.

6.4 PRESENT APPARATUS

A test section has been designed and built to give Mach 4 flow with two-dimensional expansion only[36]. This is manufactured from non magnetic materials and the internal surfaces are electrically non conducting. This rectangular section was chosen so that the whole test section could be placed between the poles of a large electromagnet (Plate 3). The interelectrode distance can be varied from zero to 7.5 cms. and B can be altered to a maximum value of 0.75 webers/m^2. The expected range of velocity of the test gas can be computed from a knowledge of the stagnation conditions and assuming isentropic expansion through the nozzle. Alternatively it can be found from the static temperature and pressure by the use of Real Gas data. In the Mach 4 test section, the velocity range will be from 2300 m/sec. to 4000 m/sec. if scramjet combustor conditions are reproduced in the shock tunnel.

Fig. 14 shows the theoretical values calculated from Eq. 8, of the induced voltage per cm., as a function of velocity, for three values of B.

If the maximum interelectrode distance is used (D = 7.5 cms) then from this graph it can be seen that the induced voltage is likely to be approximately 150 volts.

6.5 RESULTS

Current results at Sheffield show that at supersonic combustor conditions, the electrode voltage in the absence of a magnetic field is less than 0.1 volts, which at the maximum field corresponds to a minimum D of 0.25 cms., for 2% accuracy. Thus reasonably accurate velocity profiles should be obtainable with the present arrangement, and if some means can be found to elliminate the spurious voltage, even higher resolution should be possible. Preliminary results using a magnetic field confirm that the output voltage is two orders of magnitude greater than the spurious voltage (e.g. see Fig. 15 (a)).

7.0 THE MEASUREMENT OF CONDUCTIVITY

Several methods have been used to monitor conductivity levels in shock tubes, each having been developed to suit a particular piece of apparatus. The simplest form of a conductivity meter consists of two probes held at different potentials. The voltage across them and the current flowing is recorded continuously. The passage of a medium of different conductivity causes a change in current and the unknown conductivity can therefore be found. Unfortunately the probes may upset the supersonic flow in the test section, and frequently prove noisy.

Lin et al[34 and 37] measured the conductivity of ionised argon and air by means of an electrodeless technique. A search coil surrounding the shock tube is placed upstream of a field coil. The passage of the conducting medium gives rise to an

induced voltage across the search coil due to displacement of the magnetic field. A knowledge of the velocity of the gas is required for calibration, hence this method is unsuitable for use where it is required to find the velocity.

A most promising electrodeless technique is that of Olson and Lary[38] in which the dissipation of energy in a 20 Mc/s radio frequency field is related to the conductivity of the surrounding fluid. This technique has been adopted to the test section of our shock tunnel as follows.

To avoid flow disturbance the R.F. coil (1.27 cm. x 3.18 mm) was set in a piece of perspex (3.8 cm. diameter) and the surface covered with an insulating material. By using the large diameter perspex (relative to the coil size) the effects of the brass walls, into which the probe fits, were minimised. The associated circuit is that used by Harris, the details of which are given in Ref. 36. Calibration is accomplished by means of electrolytic solutions of known conductivity and an output of about 2 volt metre/mho is obtained. This design has been used extensively at our laboratory for MHD research and is capable of measuring conductivity down to $\sim 10^{-4}$ mhos/metre at frequencies above 10 kilocycles. Although this sensitivity is not as high as would be desirable,it is sufficient for feasability studies. Results to date show that the conductivity is of the order of 10^{-3} mhos/metre (see Fig. 15 (b)) and therefore adequate

for the velocity measuring technique. There are still some doubts as to the extent by which this measurement is influenced by high enthalpy boundary layers.

8.0 HIGH SPEED GAS SAMPLING

A knowledge of the composition of the gases in the test section is essential for extensive supersonic combustion research. Two particular aspects require composition information;

a) Mixing of the driven test gas (air) with the driver gas.

b) Mixing of the fuel with the air in the combustor.

These two aspects may be investigated independently and although various optical and spectroscopic methods are applicable, gas sampling and analysis provides a valuable technique. Since the running time of the tunnel is only a few milliseconds, the gas sampling process must be particularly rapid. A fast acting solenoid valve (developed by B.P.) with a sampling time of 0.5 milliseconds is available, and this period is sufficiently short for our purpose. The valve is triggered electrically by suitable delay circuits controlling the discharge of a 300μF condenser charged to 200 v. via a thyratron. The solenoid valve motion is monitored by a capacity transducer mounted on the valve stem. The gas sample of about 0.5 ml. may then be analysed by a gas chromatograph using argon as the carrier gas. Hydrogen, helium, oxygen and nitrogen may be detected with a hot wire catharometer.

(Water is absorbed in the column). It follows therefore that the concentration of all the gases of interest can be determined.

The design of the sampling head is particularly important and a supercritical 'intake' design is proposed. (See Fig. 16). The time constant of the sampling chamber must be small compared to the running time of the tunnel, so that the areas of the inlet and exhaust holes must be carefully calculated. This apparatus has not yet been used in the shock tunnel, however the results will be reported in due course.

9.0 HIGH SPEED CINE OPTICAL TECHNIQUES

The starting process in the test section, and the fuel/air mixing process both require a certain time to become established. These time dependent phenomena can be evaluated very conveniently by high speed cine optical techniques, using framing speeds in excess of 10,000 frames/sec. The Beckman and Whitley Dynafax camera is capable of speeds up to 35,000 frames/sec. and is therefore very suitable for this purpose. The total number of frames is 224, corresponding to a duration of 6.4 milliseconds at the maximum framing rate. Synchronization of the camera to the test period presents a problem, however since there is insufficient light from the combustion to expose the film, a separate light source must be used which can therefore be used for synchronization. A flash source, giving a square wave light output of variable duration (8-22 milliseconds), is available

and has been successfully used with the Dynafax camera in a conventional Toepler Schlieren system.

An off-axis, single pass optical system based on 15 cms. parabolic mirrors is set up to photograph events in the test section a short distance downstream of the nozzle. As an alternative to the cine Schlieren, a single exposure is often taken using an argon-spark light source having a duration of less than 1μ sec. (constructed from details given in Ref. (39)). Triggering of the light sources is accomplished from either an ionisation gauge or direct from an oscilloscope gate using appropriate delays. A mechanical shutter is arranged to close after the light source has triggered to prevent overexposure of the film due to the luminosity of the driver gases.

When the Dynafax camera is placed in the optical system, direct projection of the image on the film must be made to obtain sufficient exposure and retain good quality. During setting up the image must be observed directly on the film because of the nature of the camera optics.

10.0 CONCLUSIONS

1. The tailored interface hypersonic shock tunnel is a suitable facility for supersonic combustion research corresponding to scramjet operation between Mach 10 and Mach 20.

2. Since the combustion chamber operates at about one third of the flight Mach number, connected testing can be carried out in a

test section very similar to the conventional supersonic wind tunnel. With this technique reasonable simulation can be carried out with stagnation pressures $\geqslant$ 200 Bars and stagnation temperatures $\geqslant$ 5000°K.

3. These conditions can be realized with hot hydrogen or hot helium driving air at shock Mach numbers in the range 7 to 13. Operation above tailoring will result in curtailment of testing time due to contact surface instability. A driven tube length of about 10 metres is required to obtain sufficient running time for combustion experiments.

4. The dimensions of the connected test section are determined by (a) mixing and kinetic processes which indicate that 1 metre is a convenient length, and (b) friction effects which indicate that the maximum length/diameter ratio should be about 10.

5. The starting phenomena in the test section require

a) a starting pressure ratio about an order of magnitude greater than the running pressure ratio for rapid starting, and

b) pre-evacuation of the test section to about 1/30th of the safe operating pressure to prevent damage by the starting shock.

6. The flow is expected to freeze in the nozzle just before entering the test section resulting in an above equilibrium concentration of oxygen atoms.

7. A solenoid valve or electrically ruptured diaphragm can be

used to introduce fuel into the combustion test section at the required instant. Detailed knowledge of the flow field is required for satisfactory interpretation of performance.

8. The measurement of gas velocity by electromagnetic induction has been investigated. An 18 cm. diameter magnet (0.75 Wb/m^2) placed around the non-magnetic test section has been used to produce Faraday voltages proportional to velocity. An output in excess of 20 volts/cm. can be generated.

9. This technique is applicable above, (a) equivalent flight Mach numbers of 11 without seeding, (b) equivalent flight Mach numbers between 5 and 7 with seeding.

10. Electrical conductivity of the gas has been investigated using an electrodeless radio frequency probe technique. The results confirm that the gas in the test section is sufficiently conductive to permit the use of a load impedence of $>10^8$ ohms in the velocity measuring instrument.

11. A high speed gas sampling valve (0.5 milliseconds sampling time) can be used to study the concentration of species in the test section.

12. High speed (35,000 frames/sec) cine photography can be used for time resolved flow visualization techniques.

NOMENCLATURE

A	area
a	semi-width of test section (in field direction)
B	magnetic field strength
b	semi-width of test-section (perpendicular to field direction)
C_f	skin friction coefficient
D	interelectrode distance
e	charge of the electron
h	enthalpy
K_D	process efficiency
k	Boltzmann's constant
M	Mach number
P	stagnation pressure
p	static pressure
q	dynamic pressure
R_g	gas resistance
R_L	load resistance
T	stagnation temperature
t	static temperature
V	velocity
V_i	ionisation potential
v	voltage
α	electron-atom collison cross-section
ε	exponent in the pressure area relationship defined in eqn. (7).

γ specific heat ratio

η kinetic energy efficiency

σ gas conductivity

<u>Subscripts</u>

1,3,5,7	stations defined in Fig. 2.
D	diffuser
m	mean value
N	conditions in nozzle prior to firing
S	shock
T	value in test section
y	immediately behind the nozzle starting shock

<u>Superscripts</u>

*	conditions at nozzle throat
'	values for salt solution

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FIGURES

PLATES

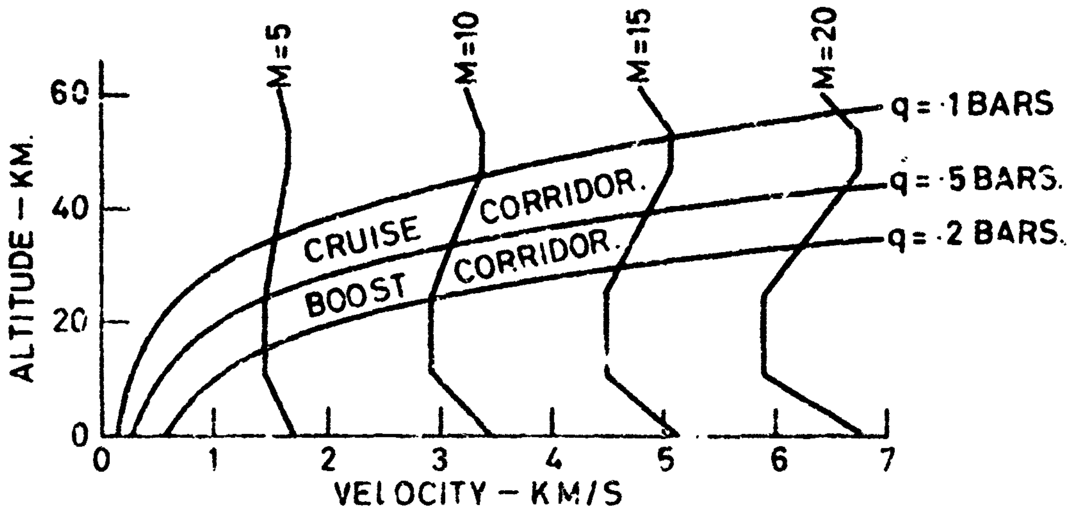


FIG. 1(a)

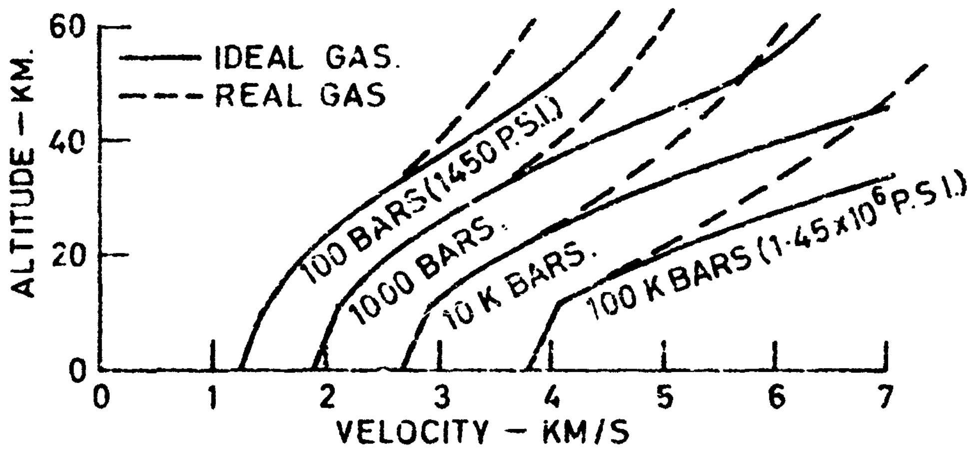


FIG. 1(b)

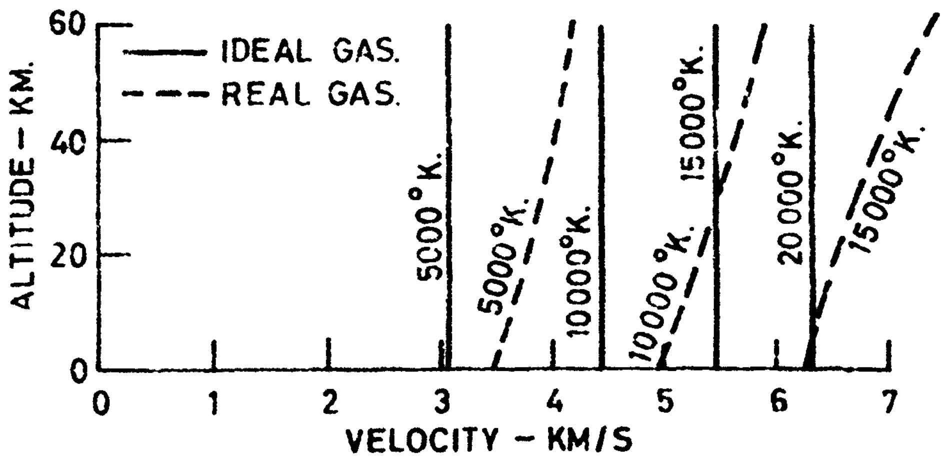


FIG 1(c)

FLIGHT STAGNATION CONDITIONS.

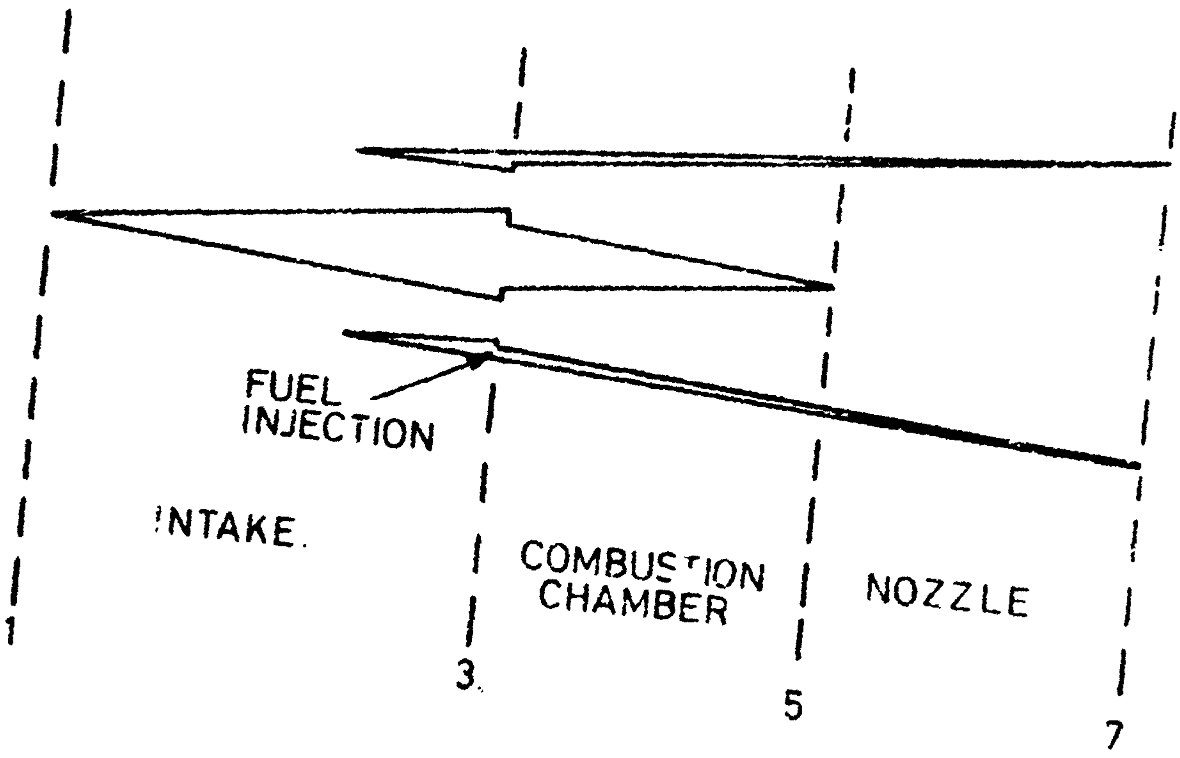


FIG. 2 ENGINE STATIONS.

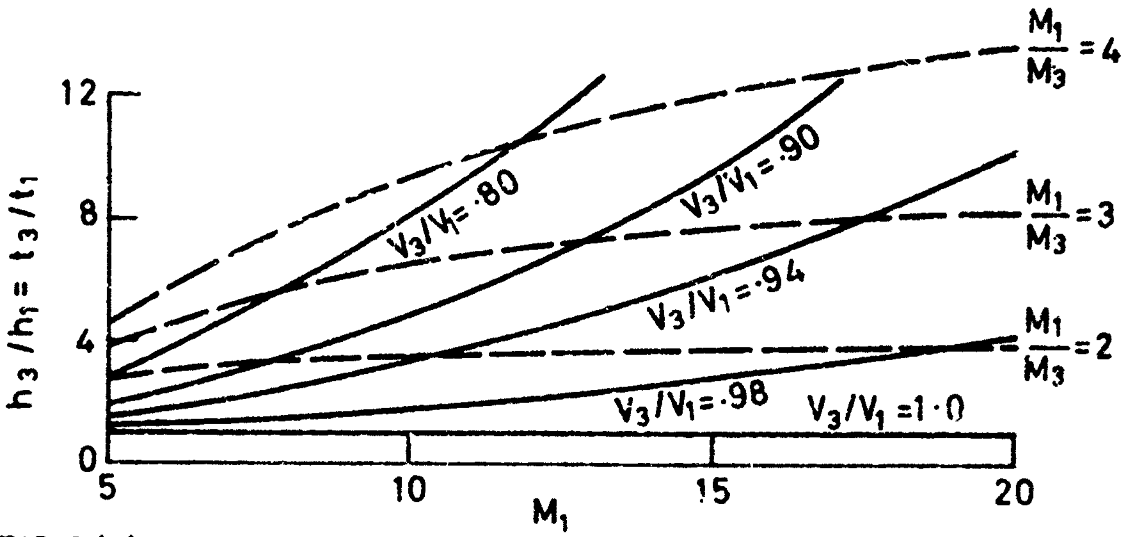


FIG. 3 (a)

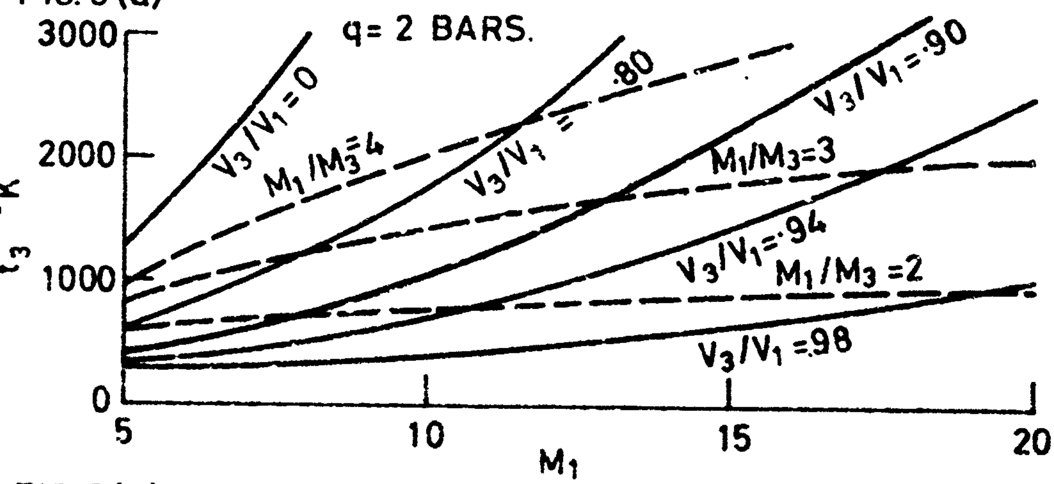


FIG. 3 (b)

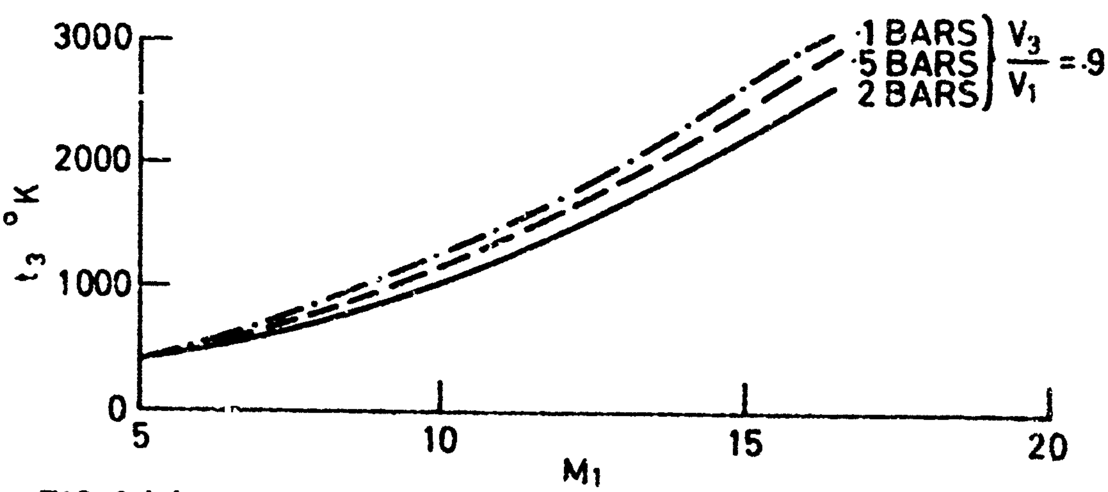


FIG. 3 (c)

COMBUSTION CHAMBER TEMPERATURES.

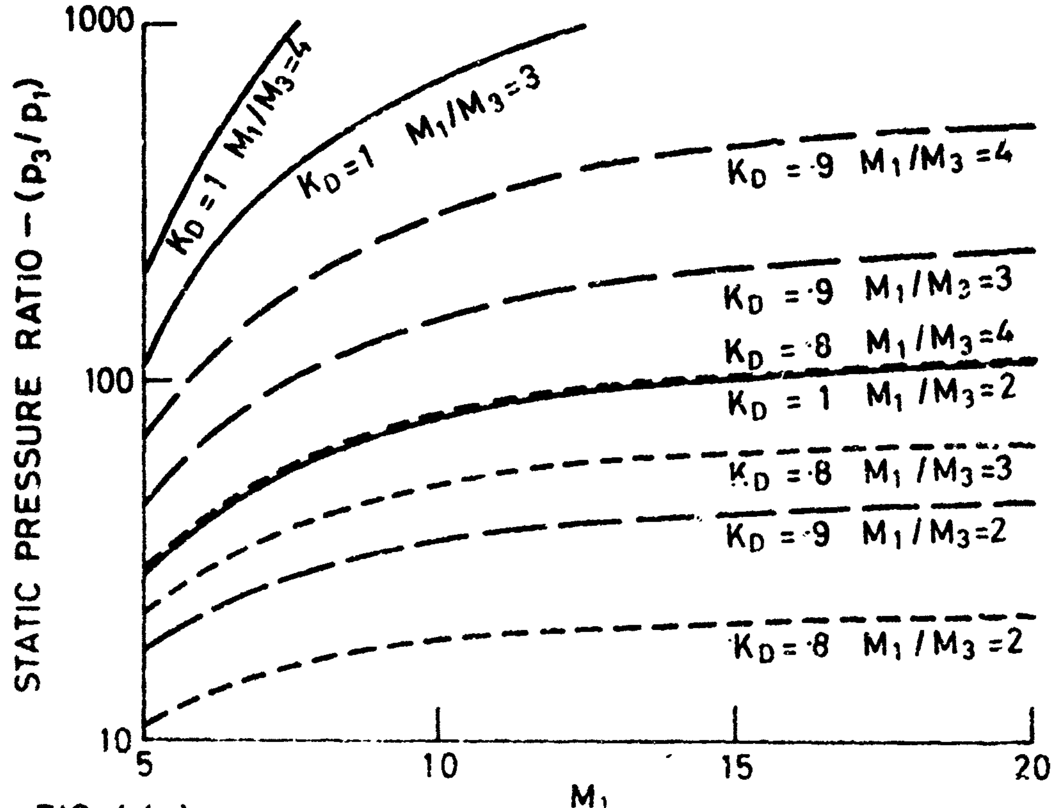


FIG. 4 (a)

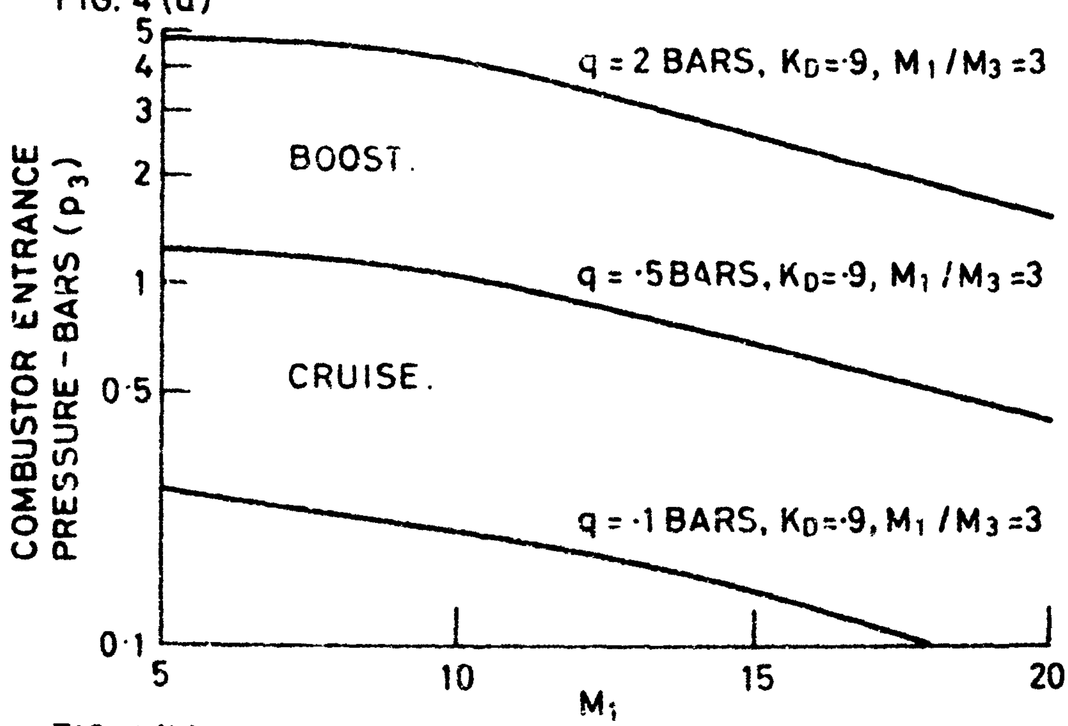


FIG. 4 (b)

COMBUSTION CHAMBER PRESSURES.

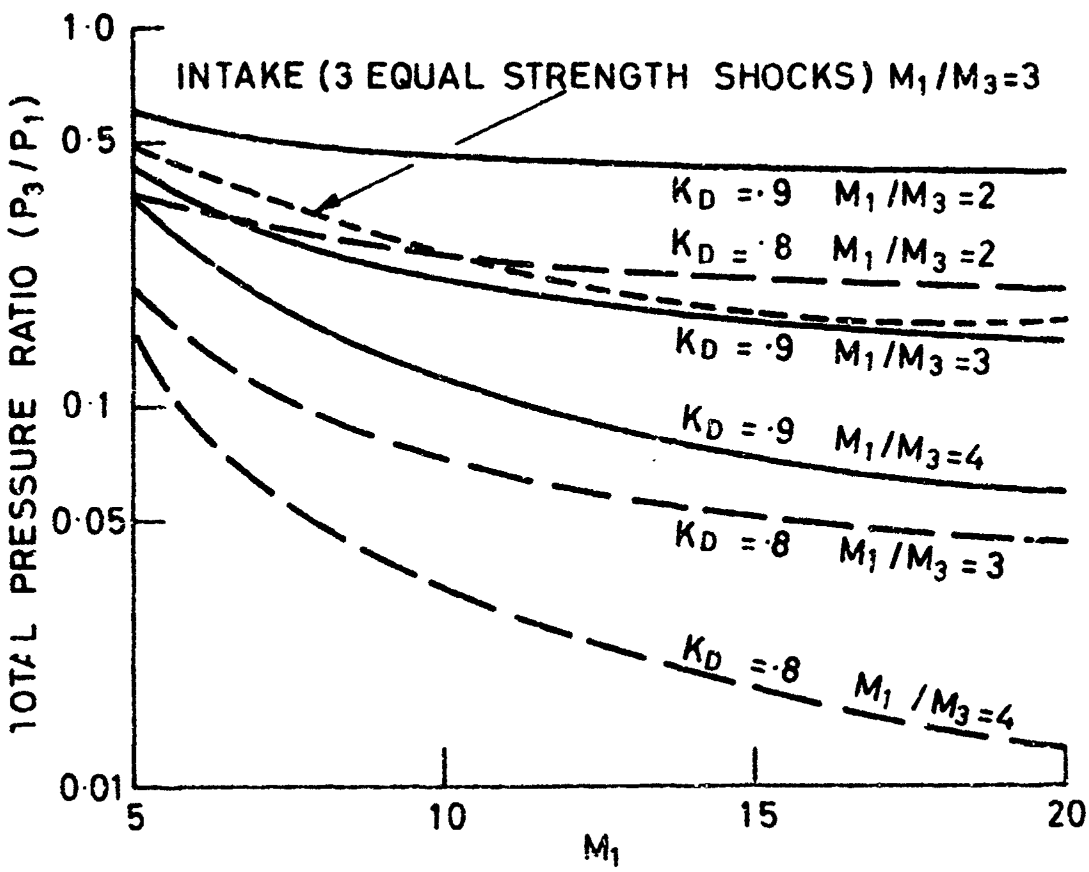
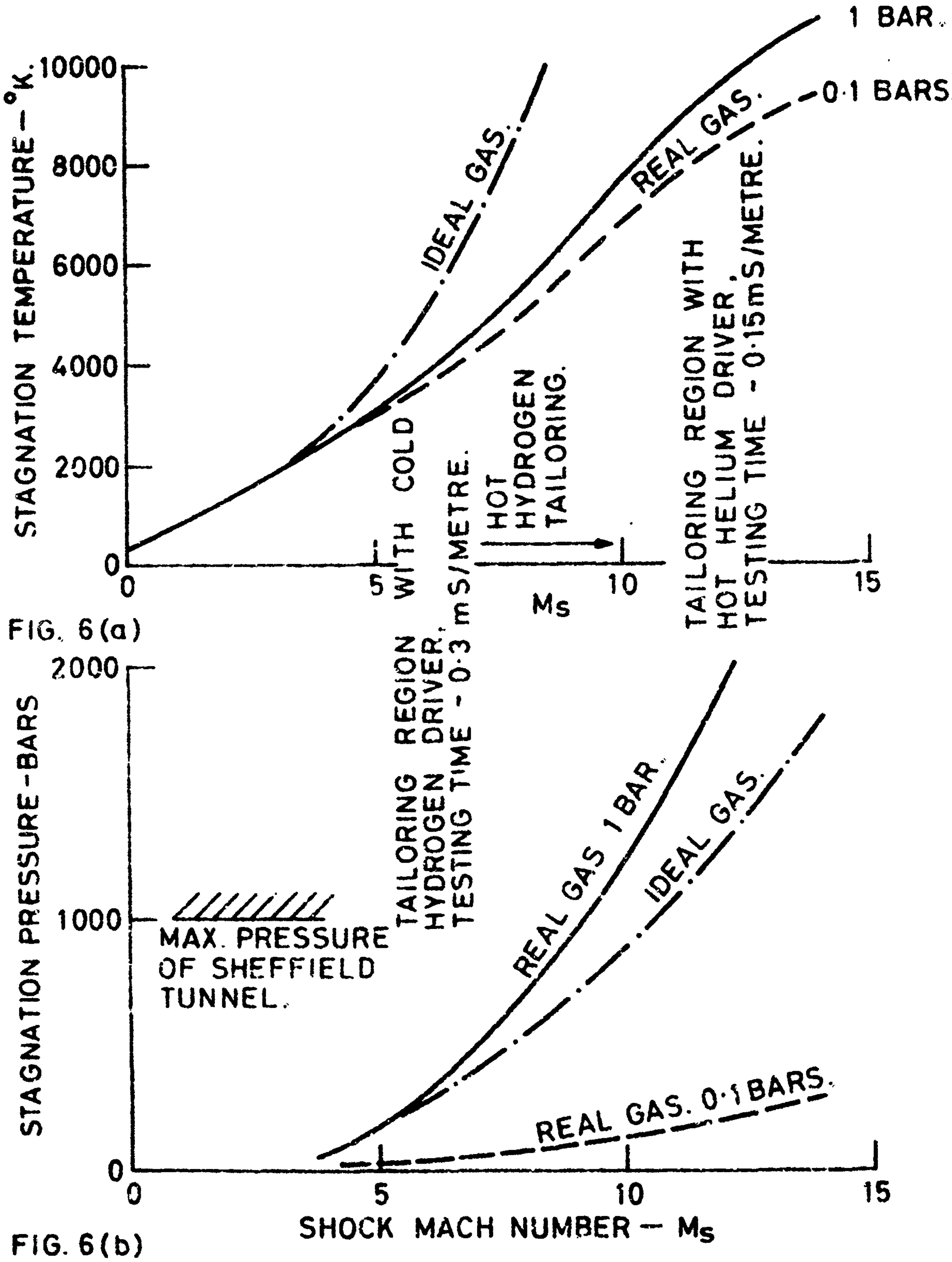


FIG.5 INTAKE PRESSURE RECOVERY.



SHOCK TUNNEL STAGNATION CONDITIONS.

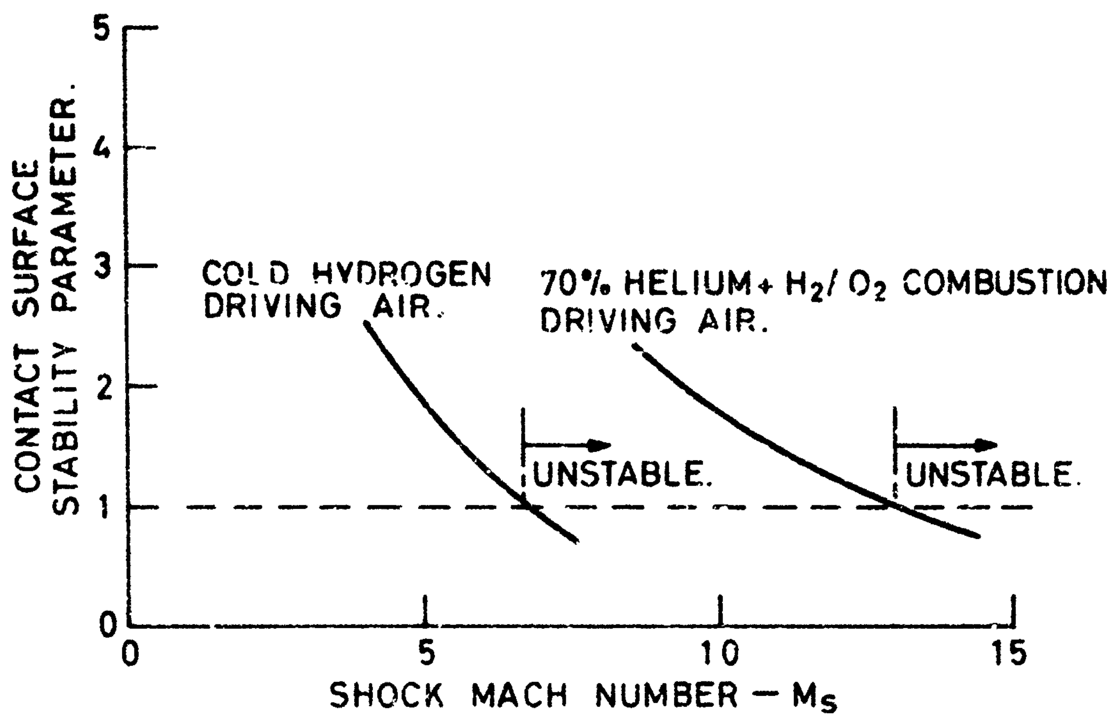


FIG. 7(a)

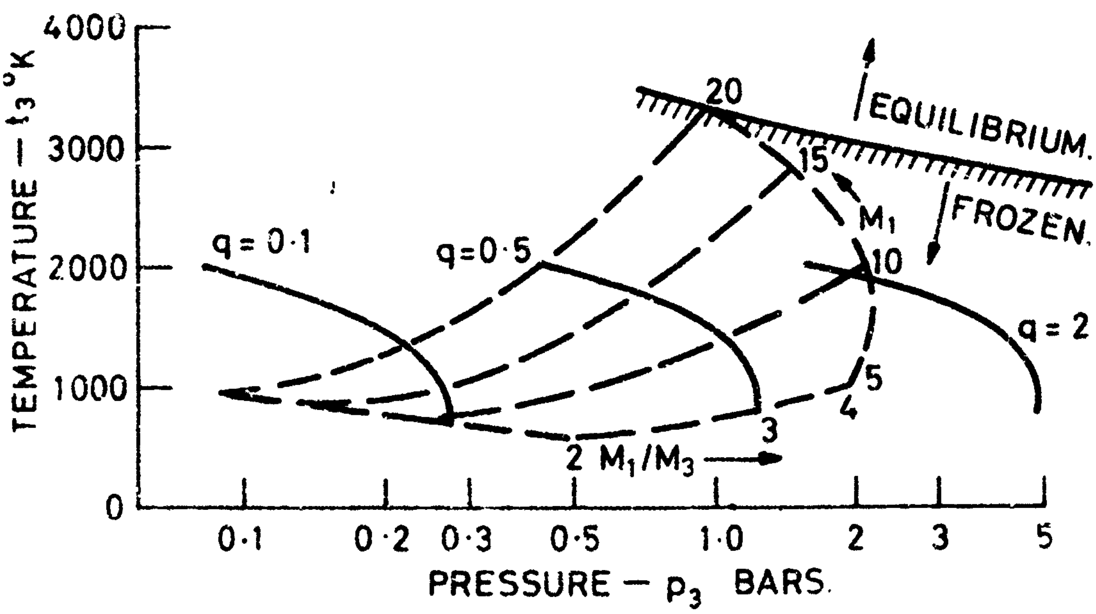


FIG. 7 (b)

CONTACT SURFACE INSTABILITY.

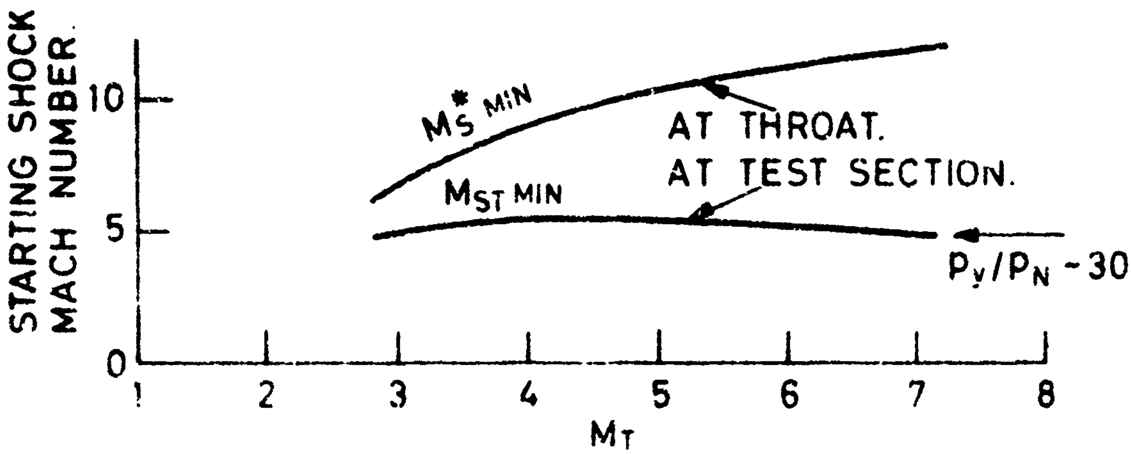


FIG. 8 (a)

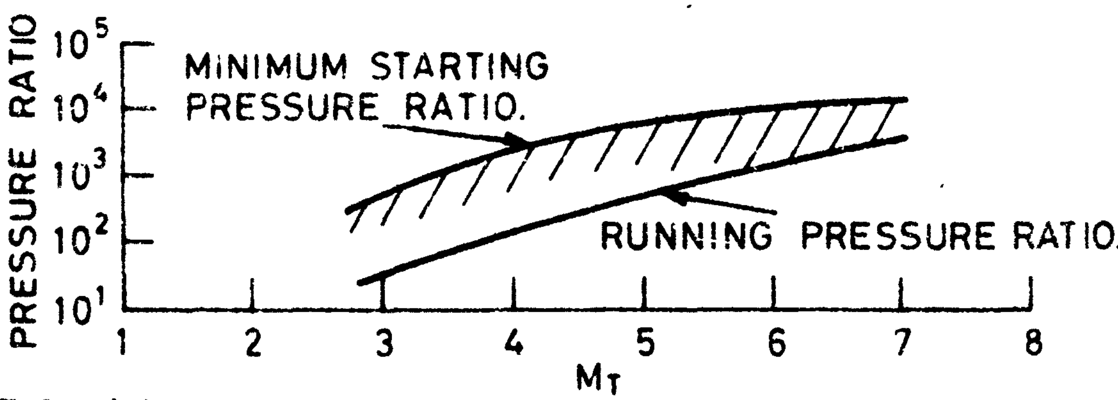


FIG. 8 (b)

TUNNEL STARTING.

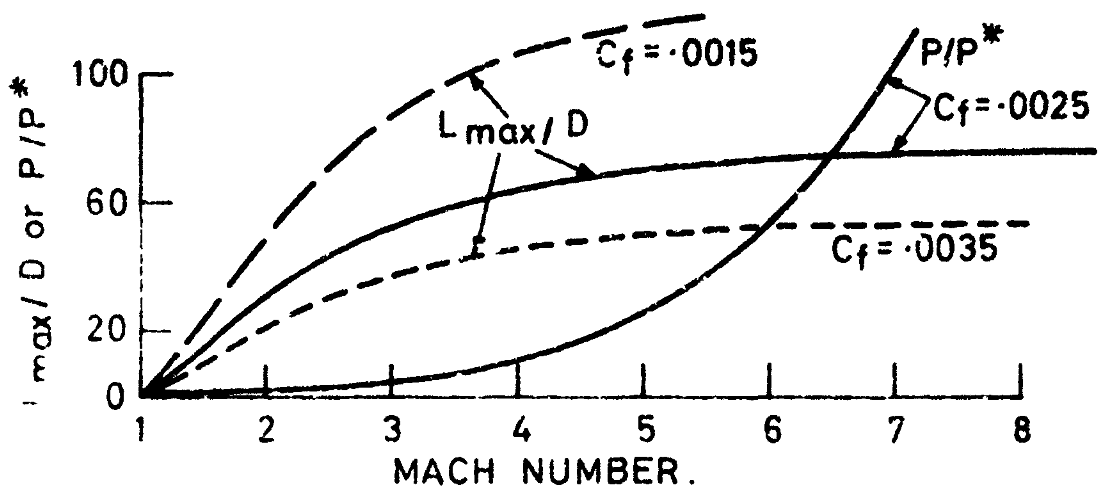


FIG. 9 (a)

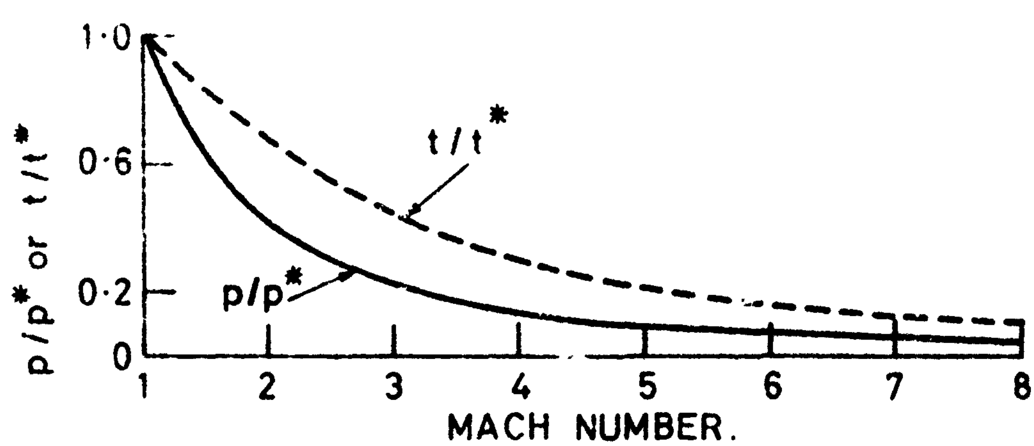


FIG. 9 (b)

FRICTION EFFECTS.

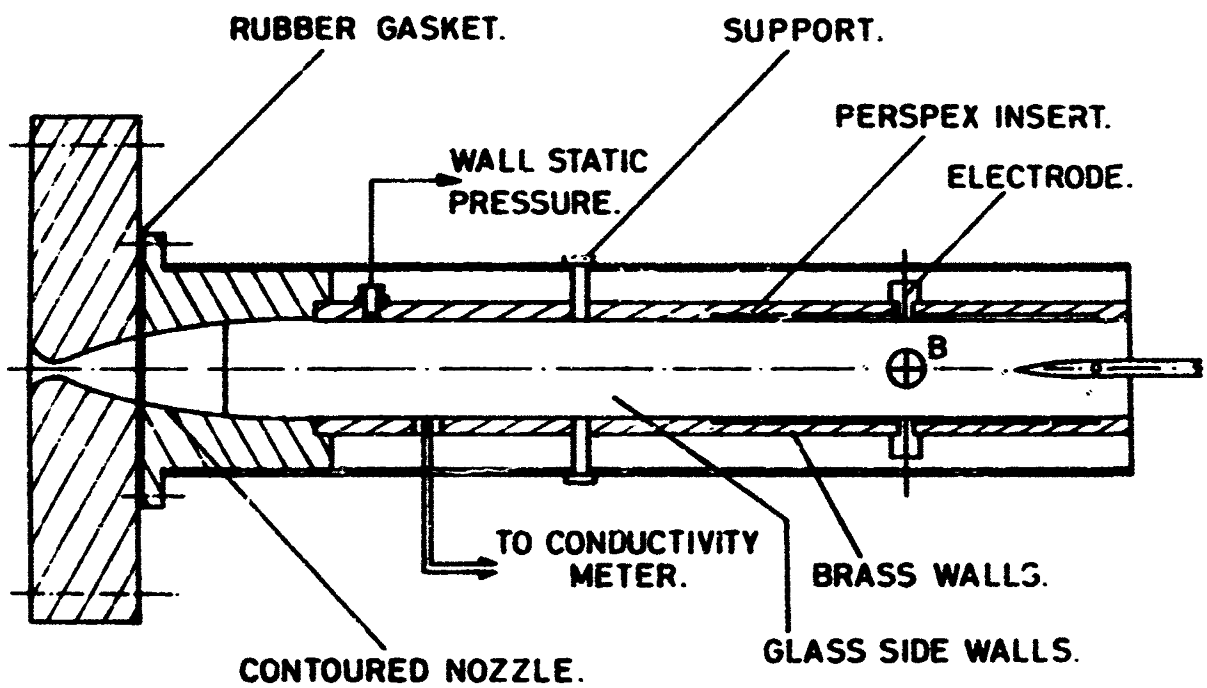


FIG. 10

MACH 4 TEST SECTION.

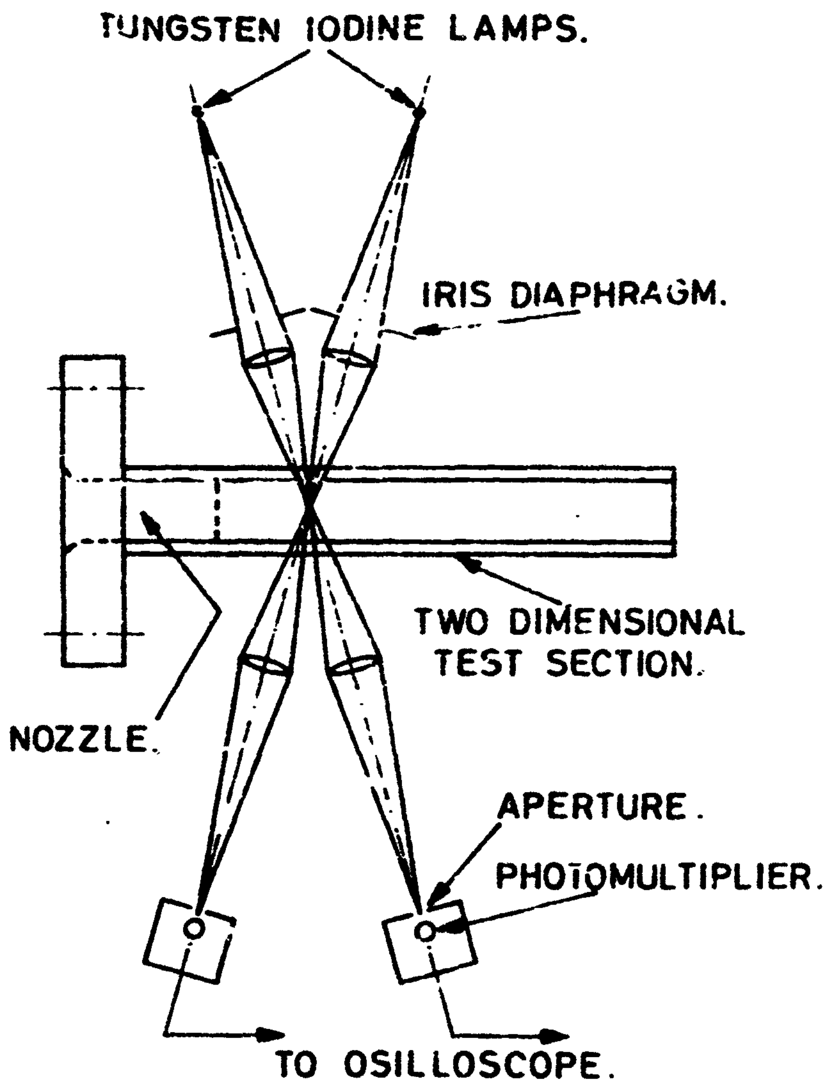


FIG. 11

DOUBLE BEAM SODIUM LINE REVERSAL APPARATUS.

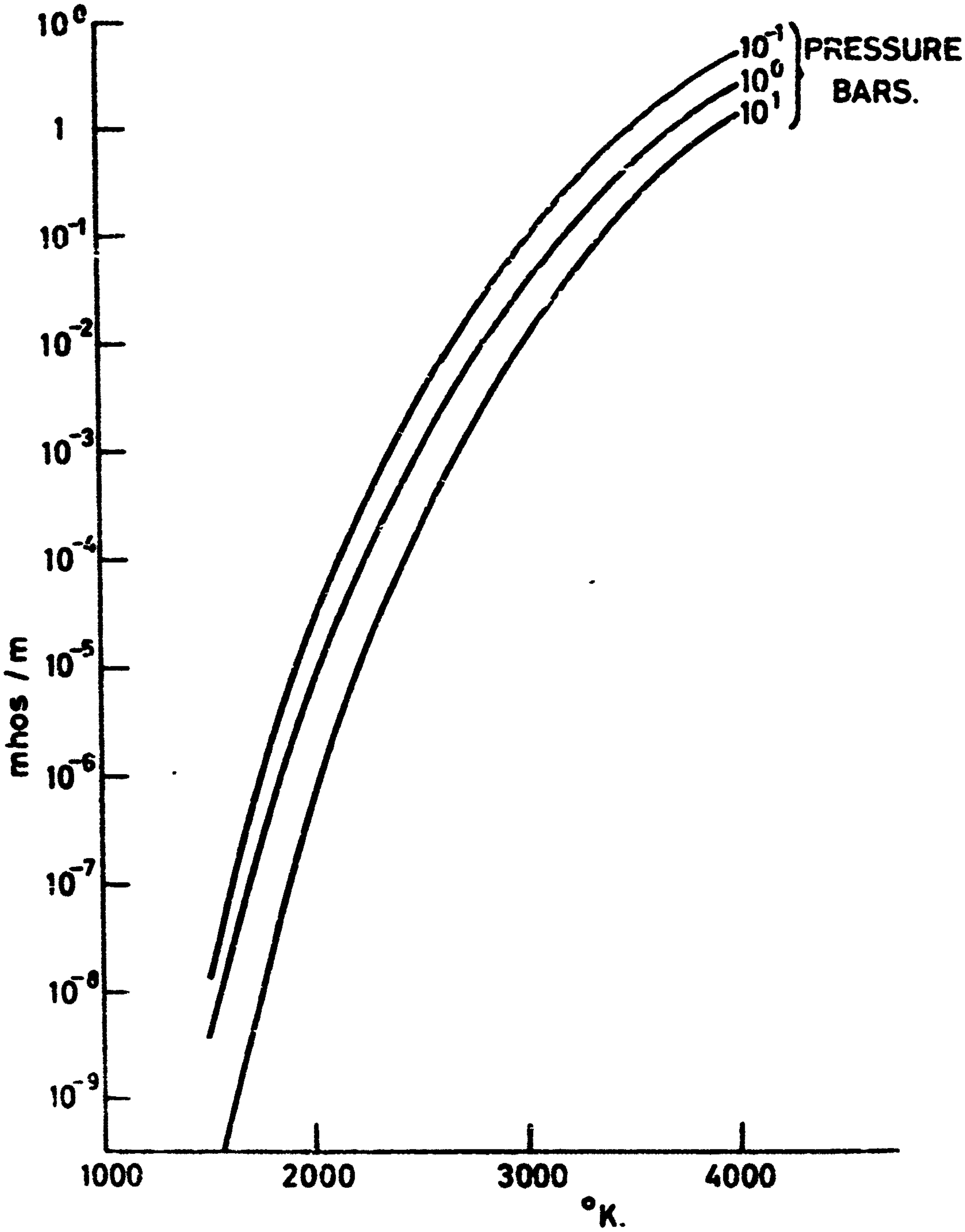


FIG. 12

CONDUCTIVITY OF AIR.

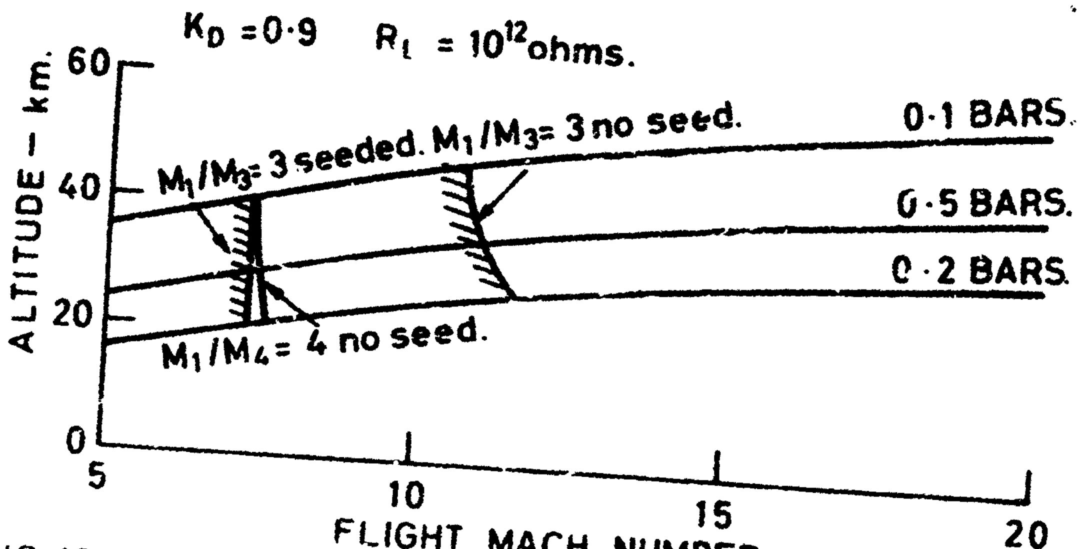


FIG. 13 LIMITS OF INDUCTION FLOWMETER APPLICABILITY.

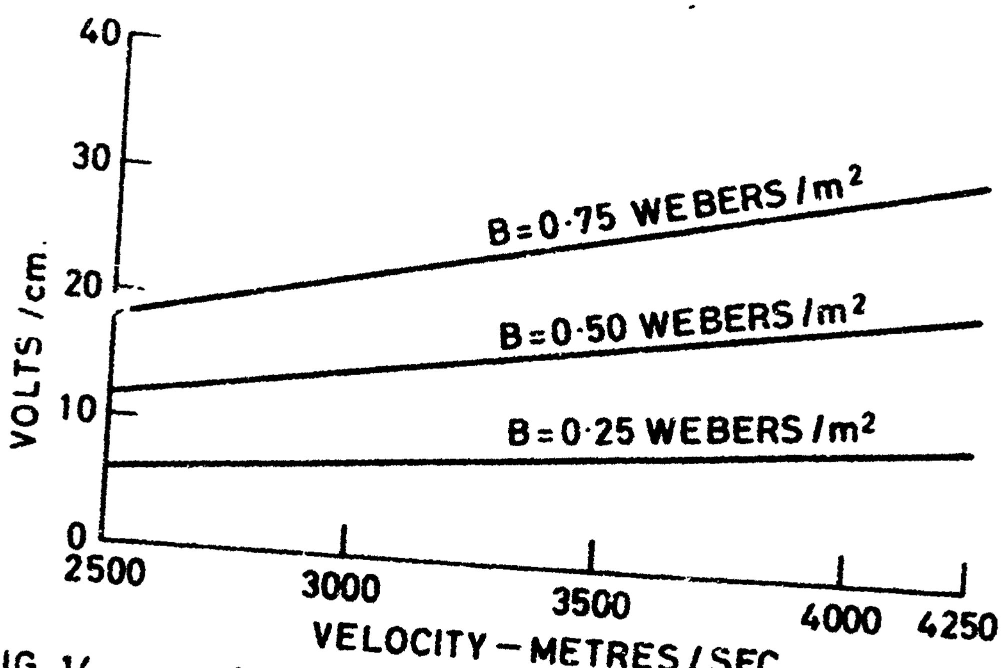


FIG. 14

INDUCED VOLTAGE VERSUS VELOCITY.

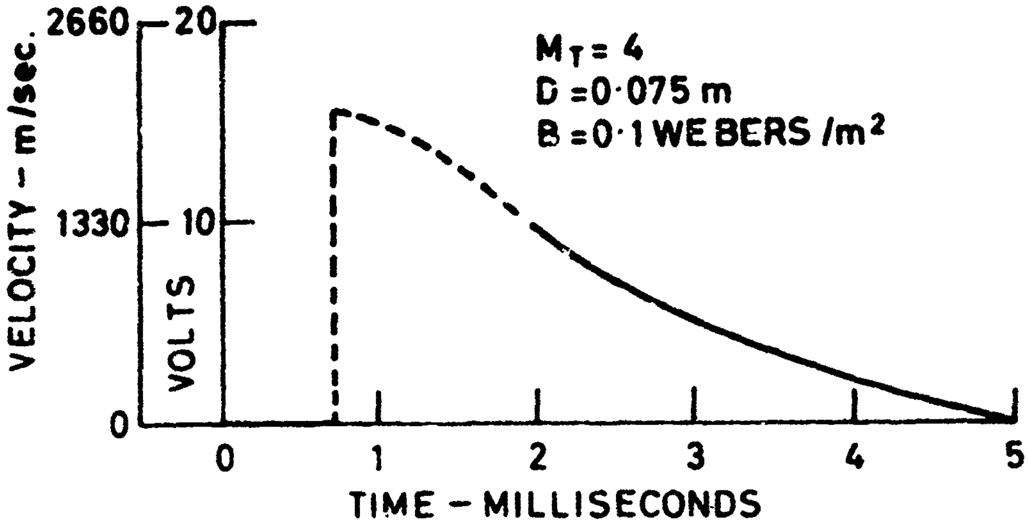


FIG 15 (a)

INDUCED VOLTAGE TRACE.

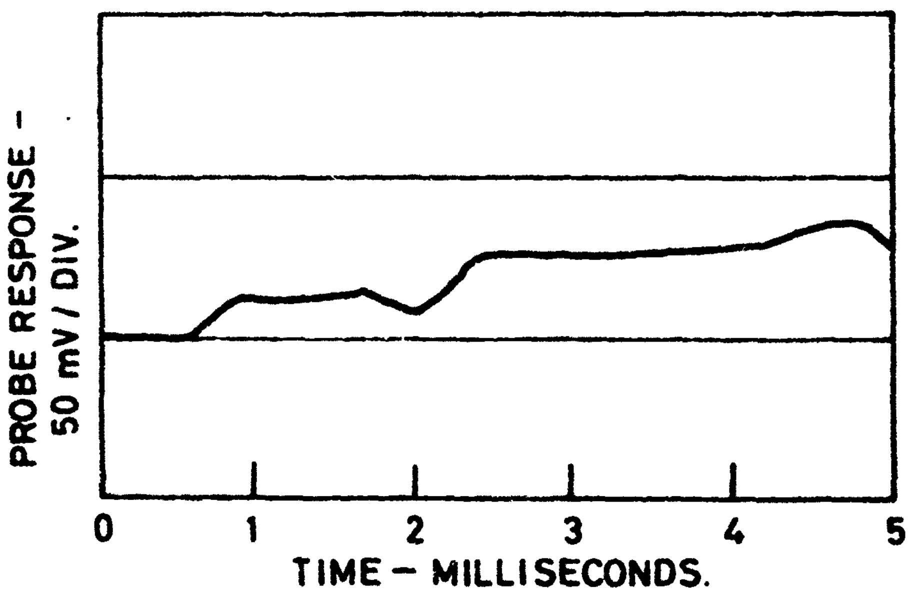
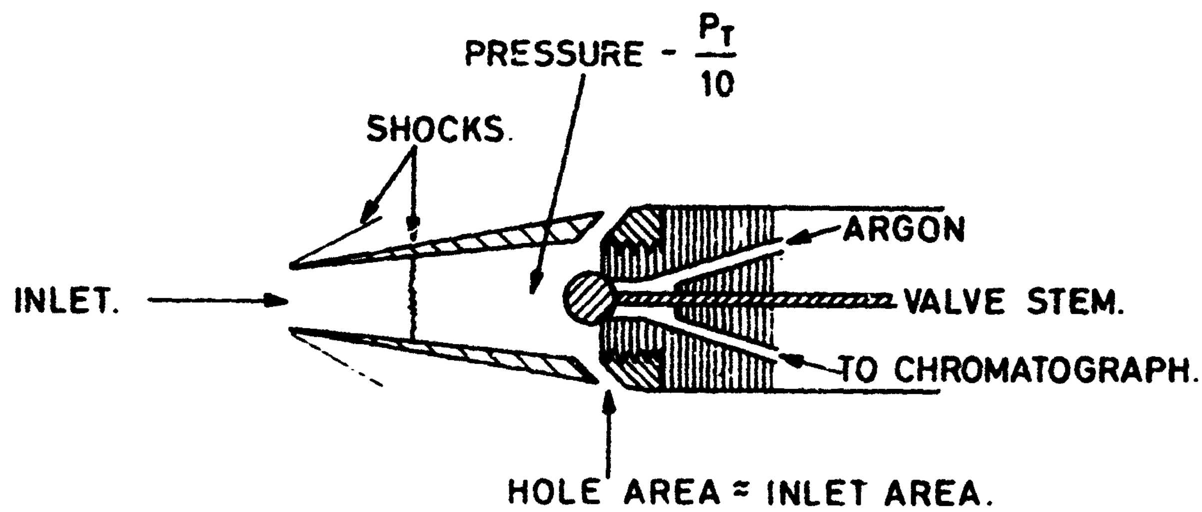


FIG. 15 (b)

CONDUCTIVITY METER TRACE.



GAS RESIDENCE TIME ~ 0·05 m. SECONDS.

FIG. 16

SAMPLING PROBE HEAD.

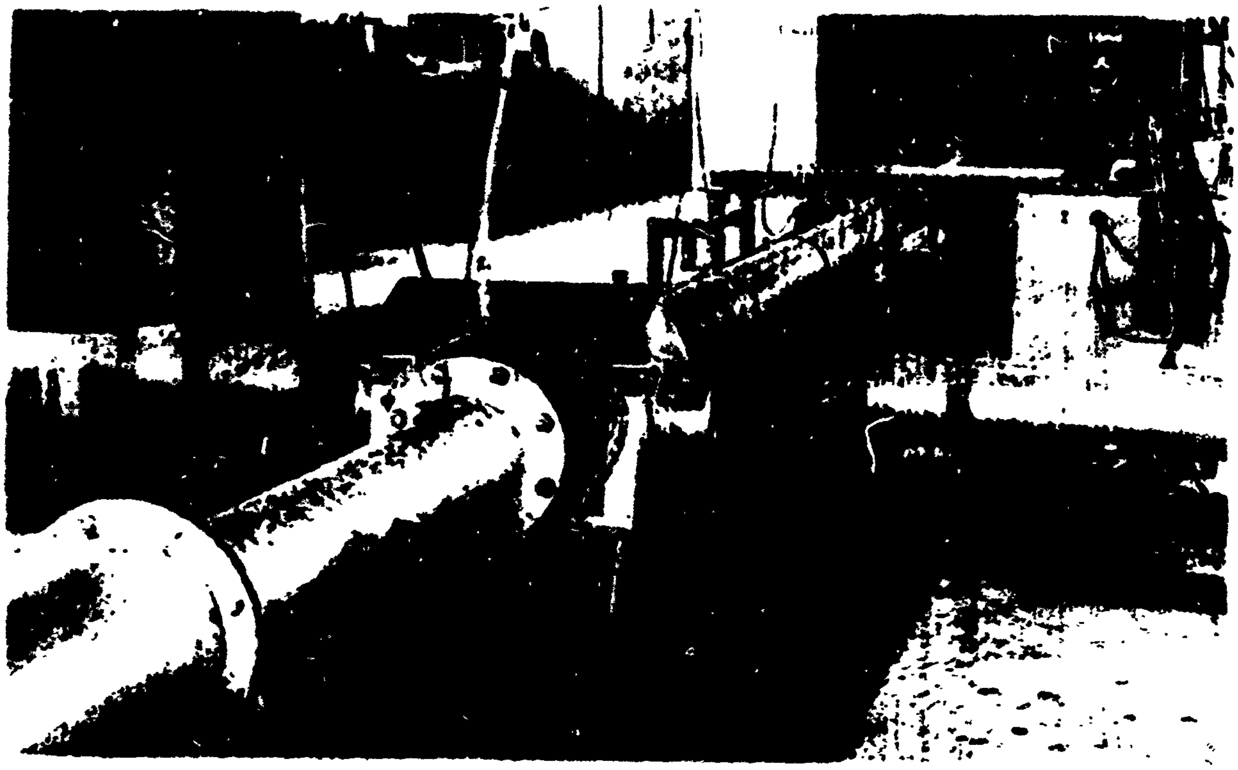


PLATE 1. GENERAL VIEW OF SHOCK TUNNEL.

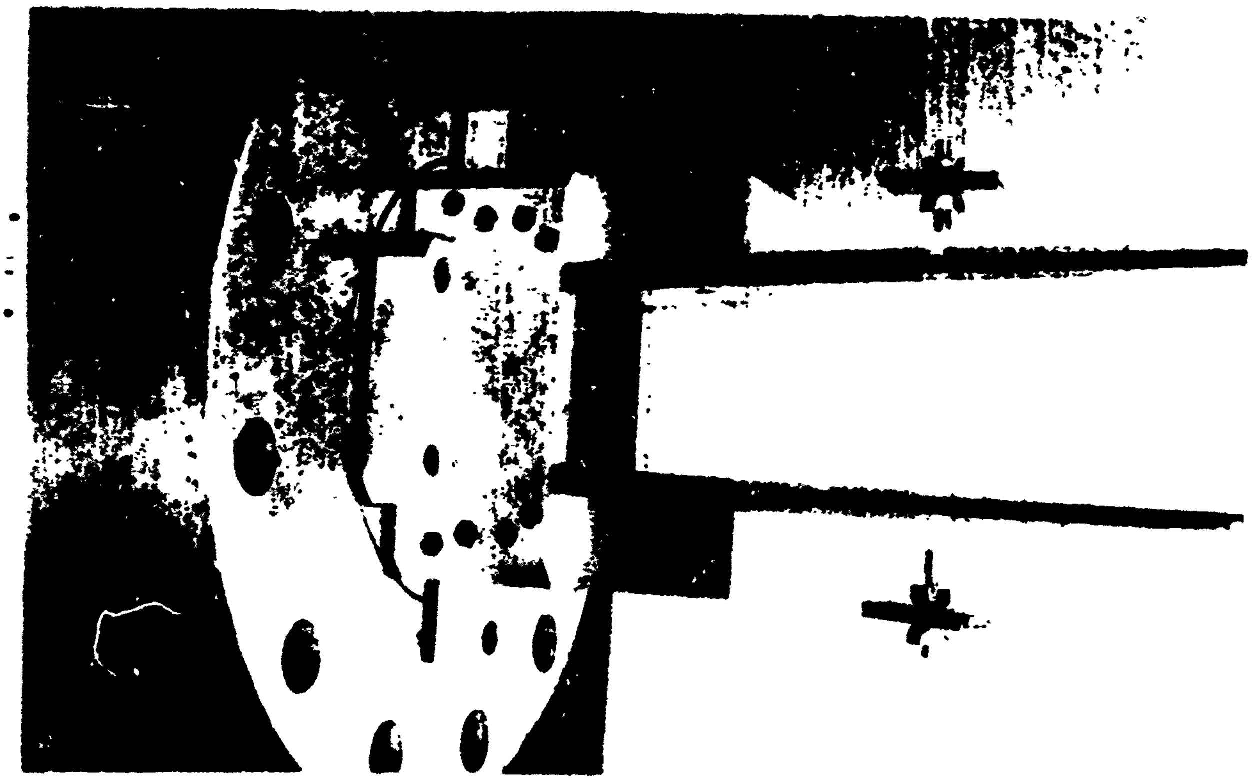


PLATE 2. MACH 4 NOZZLE ESSEMBLY.

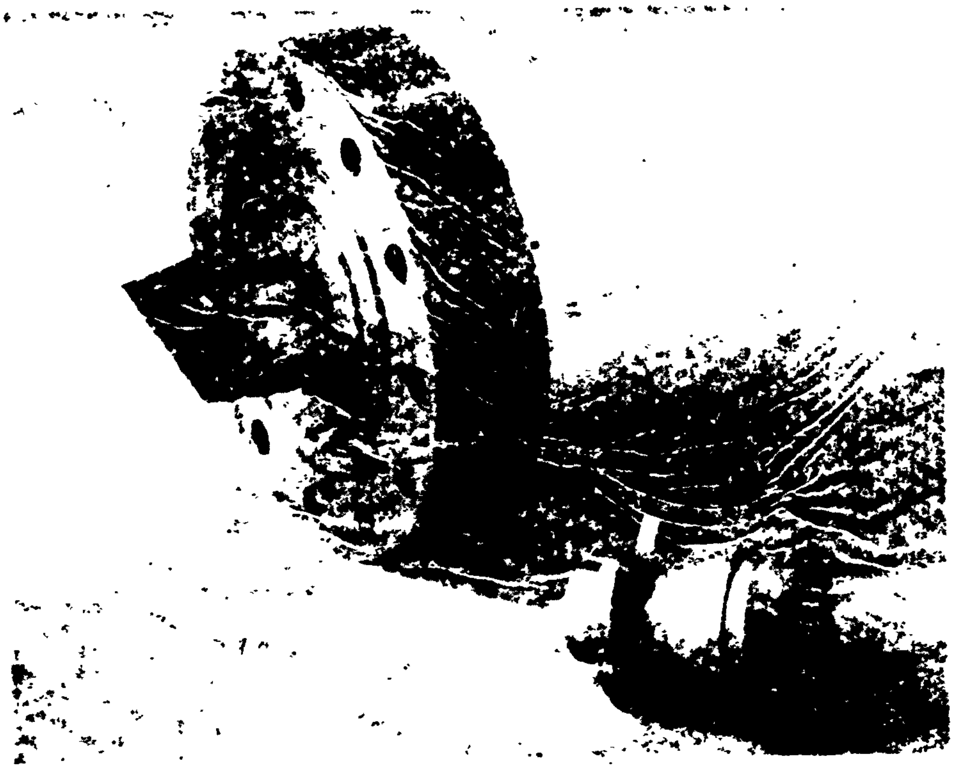


PLATE 3. INJECTOR, THROAT, AND NOZZLE EXPANSION BELL.

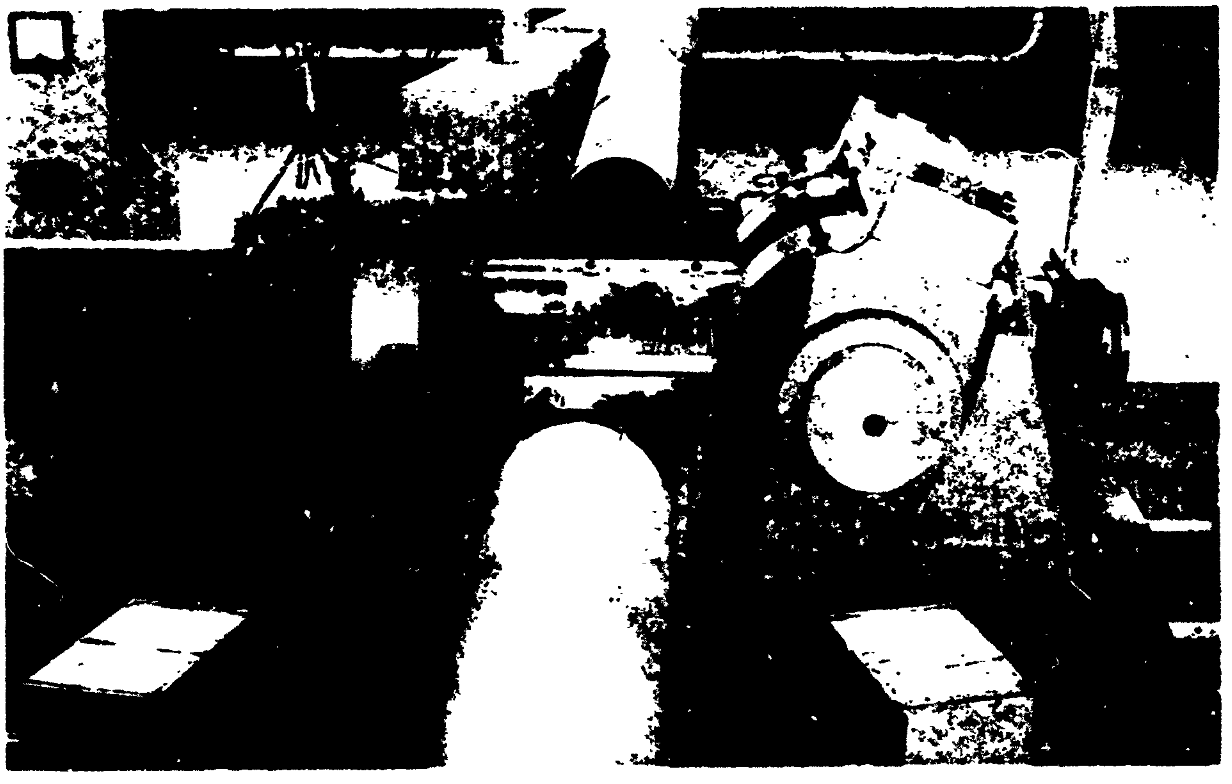


PLATE 4. TEST SECTION BETWEEN POLES OF ELECTROMAGNET.

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13. ABSTRACT

The flight corridor for scramjet operation extends to about Mach 20 at altitudes up to 50,000 metres. Simulation of the combustion chamber conditions for research and development therefore requires air at extremely high enthalpy and pressure. The tailored interface shock tunnel is an economic test facility which can attain the required conditions, despite the short testing time. The supersonic combustion test chamber may be directly connected to the shock tunnel nozzle. Starting of the flow in the test section is accompanied by a transient shock whose strength may be reduced by pre-evacuation. Fuel (usualy hydrogen) is injected into the test section and the mixing, reaction and aerodynamic processes are investigated. In addition to conventional pressure and shock speed instrumentation, the feasibility of the measurement of gas velocity by electromagentic induction has been studied and shown to be a promising technique at the higher speeds. The electrical conductivity of the gas limits the range of applicability and radio frequency probes have been used to indicate its magnitude. High speed gas sampling values may be used to measure the fuel concentration in the mixing region, however care must be taken with probe design. The proven value of flow visualization techniques may be attained by high speed cine photography.

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Shock Tunnel						
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